70p.

N64-17695

CODE-1

CR-55926

REPUBLIC
AVIATION CORPORATION

RAC 1333, Vol. 1; ~~(SSD-1027)~~ SSD-1027, Vol. 1)
17 June 1963
open. with
70 p refs

OTS: $6.60 ph, $2.30 mf

(NASA CR-55926;)

SYNCHRONOUS METEOROLOGICAL SATELLITE (SMS) STUDY

Volume 1;

Summary and Conclusions

Final Report, Feb. - May 1963

RAC 1333

Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
Goddard Space Flight Center
Greenbelt, Maryland

under
(NASA Contract ~~No.~~ NAS 5-3189)

OTS

1948322

REPUBLIC AVIATION CORPORATION
Farmingdale, L.I., N.Y.

## FOREWORD

This final report on Contract NAS 5-3189 is presented by Republic Aviation Corporation to the Goddard Space Flight Center of the National Aeronautics and Space Administration and consists of the seven volumes listed below. The period of the contract work was February through May, 1963.

The sub-titles of the seven volumes of this report are:

1 Summary and Conclusions

2 Configurations and Systems

3 Meteorological Sensors

4 Attitude and Station Control

5 Communications, Power Supply, and Thermal Control

6 System Synthesis and Evaluation

7 Classified Supplement on Sensors and Control

Except for Volume 7, all of these are unclassified. Volume 7 contains only that information on specific subsystems which had to be separated from the other material because of its present security classification. Some of these items may later be cleared for use in unclassified systems.

Volumes 3, 4, and 5 present detailed surveys and analyses of subsystems and related technical problems as indicated by their titles.

In Volume 2, several combinations of subsystems are reviewed as complete spacecraft systems, including required structure and integration. These combinations were selected primarily as examples of systems feasible within different mass limits, and are associated with the boosters to be available.

Volume 6 outlines methods and procedures for synthesizing and evaluating system combinations which are in addition to those presented in Volume 2.

Volume 1 presents an overall summary and the principal conclusions of the study.

TABLE OF CONTENTS

TABLE OF CONTENTS (Continued)

LIST OF ILLUSTRATIONS

# SECTION 1 - OBJECTIVES

## A. STATEMENT OF WORK

This volume summarizes the results of a four-month study, the subject of a contract between the Goddard Space Flight Center and Republic Aviation Corporation. The results are presented more completely in the six additional volumes listed in the foreword.

The general objectives of the study were:

(1) To find the most reasonable and reliable systems for bringing maximum areas of the Earth under constant observation

(2) To identify the critical scientific and engineering problem areas and the advances in technology required

(3) To review possible system trade-offs and alternatives, primarily to study their effects on the quality of meteorological data outputs.

In order to obtain the most reasonable and reliable systems, the approach used was to study subsystems parametrically in order to have, as stated in the contract, "a matrix of spacecraft parameters in which the meteorological sensors form the basic datum line." With the basic subsystem data available, the next step was to combine subsystems into concepts of overall systems. The system concepts were then considered in the light of the available launch vehicle payload capabilities: 100 lb for the Thor-Delta, 500 lb for the Atlas-Agena, and 1,000 lb for the Atlas-Centaur.

A direct result of the system and subsystem study was the identification of problem areas. In general, the problems are of a straightforward engineering type which can be solved by presently known techniques. No new scientific developments beyond the present state of the art are required. Feasibility limits were set for primary items, such as those concerned with the sensor data system and the accuracy required of the attitude control system.

In the study of the systems, trade-offs and alternative designs were investigated. The objective of the alternate designs was to obtain optimum quality in the sensor data in a spacecraft of reasonable size and simple in operation.

The interface characteristics for the following functions were considered as constraints for the spacecraft:

(1) Boosting the satellite to orbit altitude

(2) Injecting the spacecraft into the desired orbit at approximately the desired station

(3) Tracking, computation, and guidance for ascent and orbit injection

(4) Tracking and computation for station keeping after final orbit is attained

(5) The preparation of "processed" meteorological data for transmission to the spacecraft for relay to ground stations.

## B. SYSTEM CONCEPT

The specific SMS system which was studied consisted of one satellite, one control and data acquisition (CDA) station, and additional ground stations, equipped with simpler equipment than that in the CDA station, capable of receiving meteorological data from the spacecraft. These latter stations are referred to as multiple data acquisition (MDA) stations. The satellite is "stationary" at an altitude of 22,240 miles over the equator, at 90°W longitude. The CDA station is located in North America. The MDA stations may be located at any suitable point within line of sight of the spacecraft. The principal elements, functions, communication links, and types of data to be delivered are summarized schematically in Figure 1-1.

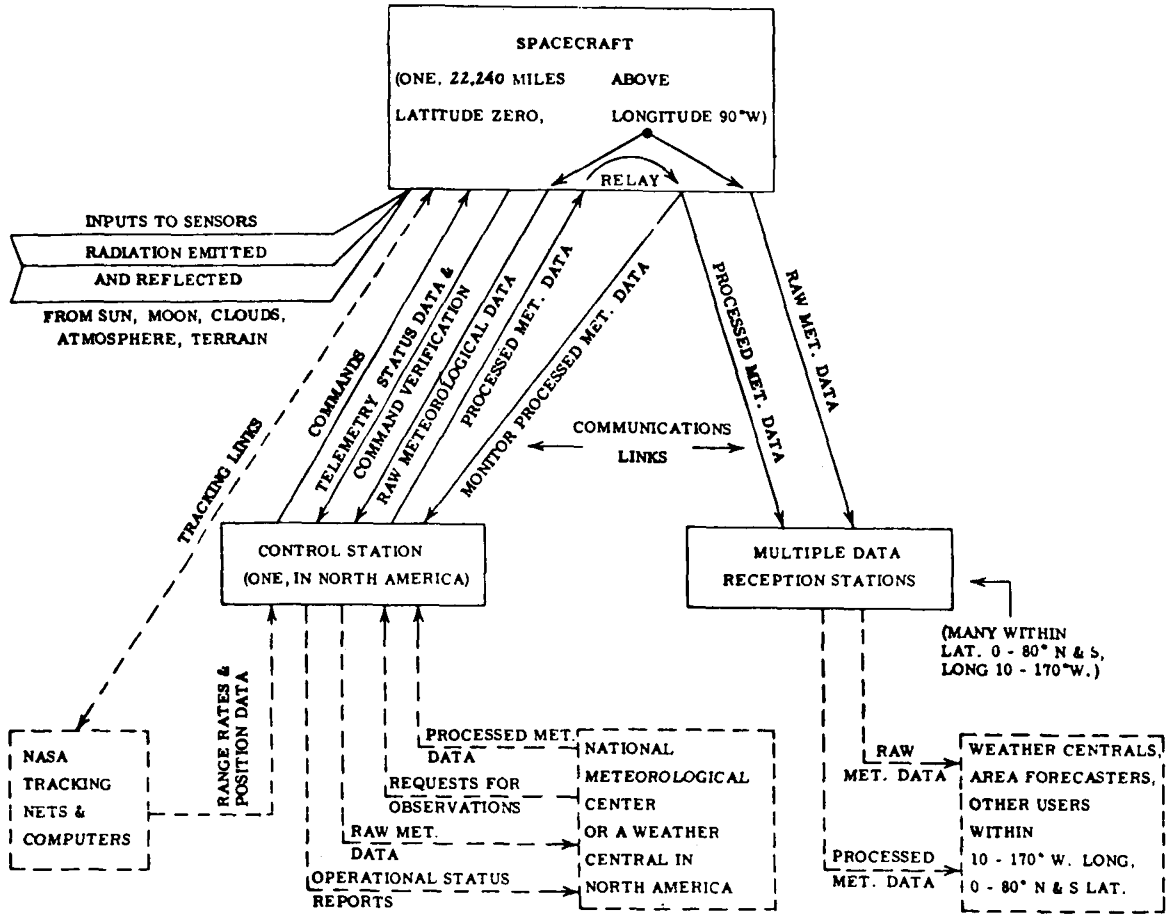



Figure 1-1. SMS System, Principal Elements and Links

The raw meteorological data to be acquired by the spacecraft and transmitted to the ground stations consist of cloud cover pictures and infrared heat budget data. The cloud cover pictures include a wide-angle picture of the entire disc of the Earth as viewed from the satellite and other narrow-angle pictures of selected smaller areas of the Earth obtained by aiming the sensor optics. Infrared measurements are obtained by scanning the Earth with a scanning spot size on the order of 100 to 300 miles. A complete cycle of meteorological data is transmitted to the ground station at least every 30 minutes. It is also desired to relay processed meteorological data from the CDA station through the spacecraft to the MDA stations.

The primary function of the CDA station is to control the operation of the spacecraft and to receive the raw meteorological data. To accomplish this function, it also tracks the satellite and monitors operation of the spacecraft by the receipt of telemetry information. Since the CDA station is in continuous contact with the SMS, there is little need for internally programmed control or monitoring functions within the spacecraft. There are consequent gains in satellite simplicity, reliability, operating life, and response to changing conditions.

The spacecraft is to be carried into synchronous equatorial orbit by a standard launch vehicle no larger than the Atlas-Centaur. The desired launch date is 1966 or later. The operating life in orbit is at least one year.

## SECTION 2 - CONCLUSIONS

The principal conclusions of this study are:

(1) It is possible, with present technology, to develop and build a synchronous meteorological satellite capable of obtaining valuable meteorological data which cannot be obtained by other methods.

(2) An SMS of about 500 lb can be put into synchronous orbit by an Atlas-Agena booster vehicle. It can provide data on the Earth's heat budget and cloud cover. In daylight areas, cloud cover resolution down to 1.3 miles per TV line is possible with frame coverage of 1250 x 1250 miles. Twenty-five such frames can cover a field at least 60 degrees of longitude or latitude from the satellite sub-point in a few minutes. In night areas, resolution becomes 4 miles. The full disc of the Earth can be covered in one frame with a resolution of 7 miles for daylight areas.

(3) To optimize the system design, it is desirable to use specific equipment for multiple functions. This results in lower power requirements, a reduction in the amount of equipment, and a lighter configuration. For example, in the specific configurations studied:

   (a) Use of the cloud pictures (both high resolution and full Earth disc, with horizon segments, landmarks, and landmark patterns) for improving location accuracy of weather features, and for monitoring and correcting the attitude control and station keeping subsystems

   (b) Use of the sensor data link as the "down-link" for relays of processed meteorological data from the CDA station to the MDA station

   (c) Use of the spacecraft equipment in this relay link with an omnidirectional antenna as an S-band transponder compatible with Goddard Range and Range Rate tracking nets, during ascent and injection

   (d) Use of the command and telemetry channels for relay of processed meteorological data of narrower bandwidth from the CDA station to Nimbus APT Ground Stations

   (e) Use of the spacecraft equipment for command and telemetry channels as a transponder compatible with the Goddard Range and Range Rate tracking nets, during ascent and injection

(f) Use of the attitude torquing jets and gas supply as thrusters to provide velocity changes for station keeping purposes.

(4) The 3-axis stabilization system, with associated configurations, is desirable in preference to other types. Reaction wheels are used for roll and pitch stabilization, with yaw stabilization provided by a constant-speed wheel operating in a gyro-compassing mode. Cold gas is used to unload the reaction wheels. Spin stabilization provides a lighter weight control system than the 3-axis system; however, the complication of providing image motion compensation for the sensors results in little or no overall spacecraft weight saving. Stabilization by gravity gradient offers some simplicity in the control system, but active damping devices are required which make the complete system equal to the 3-axis system in weight and complexity. The gravity gradient satellite also has serious developmental problems associated with deploying, separating, and orienting two connected bodies in orbit.

(5) An SMS of a size to fit the payload envelope of the Atlas-Centaur would provide, within present state of the art, resolution of 0.65 miles per TV line for daylight areas and 2 miles for dark areas. The field of view would be 625 x 625 miles at the nadir, and about 100 pictures would be required to cover the disc of the Earth. Reliability and operating life would be enhanced by the inclusion of redundant equipment for all critical functions.

(6) A configuration to accomplish minimal SMS objectives is estimated to weigh about 200 lb. This configuration would be spin stabilized. The cloud cover sensor would take one picture of the whole Earth's disc with a resolution of 7 miles per TV line. A fast shutter would be used rather than an image motion compensation mechanism. Heat budget data would be one data point for the whole Earth disc. A 12-DB loss in the sensor data channel would result from the use of a pancake beam antenna instead of an Earth-oriented antenna, as would be used in a 3-axis stabilized vehicle.

## SECTION 3 - SYSTEM STUDIES

The system study was parametric in approach and in results. Parametric data on subsystems were compiled, after which representative systems were synthesized. Then, knowing that three launch vehicles were available (Thor-Delta, Atlas-Agena, and Atlas-Centaur), an iteration was conducted to obtain a payload which would meet the SMS objectives and fit the launch vehicles. For comparison, these payloads are referred to as minimum, nominal and redundant/ nominal configurations.

The optimum configuration of the minimum capability spacecraft was one weighing 100 pounds. This could not be achieved, but a light-weight spacecraft capable of meeting minimal SMS objectives worked out at 218 pounds. The nominal spacecraft configuration weighs 524 pounds and can be orbited by an Atlas-Agena, with a margin for growth of about 100 pounds. For a larger spacecraft the approach was to enhance reliability and operating life by adding redundant units for primary subsystems and to somewhat improve sensor resolution over that of the nominal spacecraft. The configuration weighs 775 lb, which is well within the capability of the Atlas-Centaur.

Figure 3-1 illustrates the resolution levels that may be achieved under several conditions, in cloud cover pictures that can be obtained from synchronous meteorological satellites with weights of 218, 524, and 775 lb. Also shown are the weight distributions of subsystems with corresponding trend lines to indicate likely values for satellites of different orbital weights. The same weight distributions are listed in Table 3-1, with the addition of percentages associated with the principal subsystems. Additional performance capabilities of these three satellites are summarized in Table 3-2. The three configurations are shown in Figures 3-2, 3-3, and 3-4.

TABLE 3-1
WEIGHT DISTRIBUTION FOR THREE SMS CONFIGURATIONS

| | lb. | % | lb. | % | lb. | % |
|---|---|---|---|---|---|---|
| Total in Orbit | 218 | 100 | 524 | 100 | 775 | 100 |
| Meteorological Sensors | 20 | 9.2 | 83 | 15.8 | 161 | 20.8 |
| Communications, incl. Telemetry, Command, and Data Processing | 34 | 15.6 | 52 | 9.9 | 78 | 10.1 |
| Control System for Attitude and Station | 43 | 19.7 | 122 | 23.2 | 161 | 20.8 |
| Power Supply System | 60 | 27.5 | 128 | 24.4 | 172 | 22.2 |
| Thermal Control and Wiring | 13 | 6.0 | 32 | 6.1 | 46 | 5.9 |
| Structure | 48 | 22.0 | 108 | 20.6 | 157 | 20.2 |

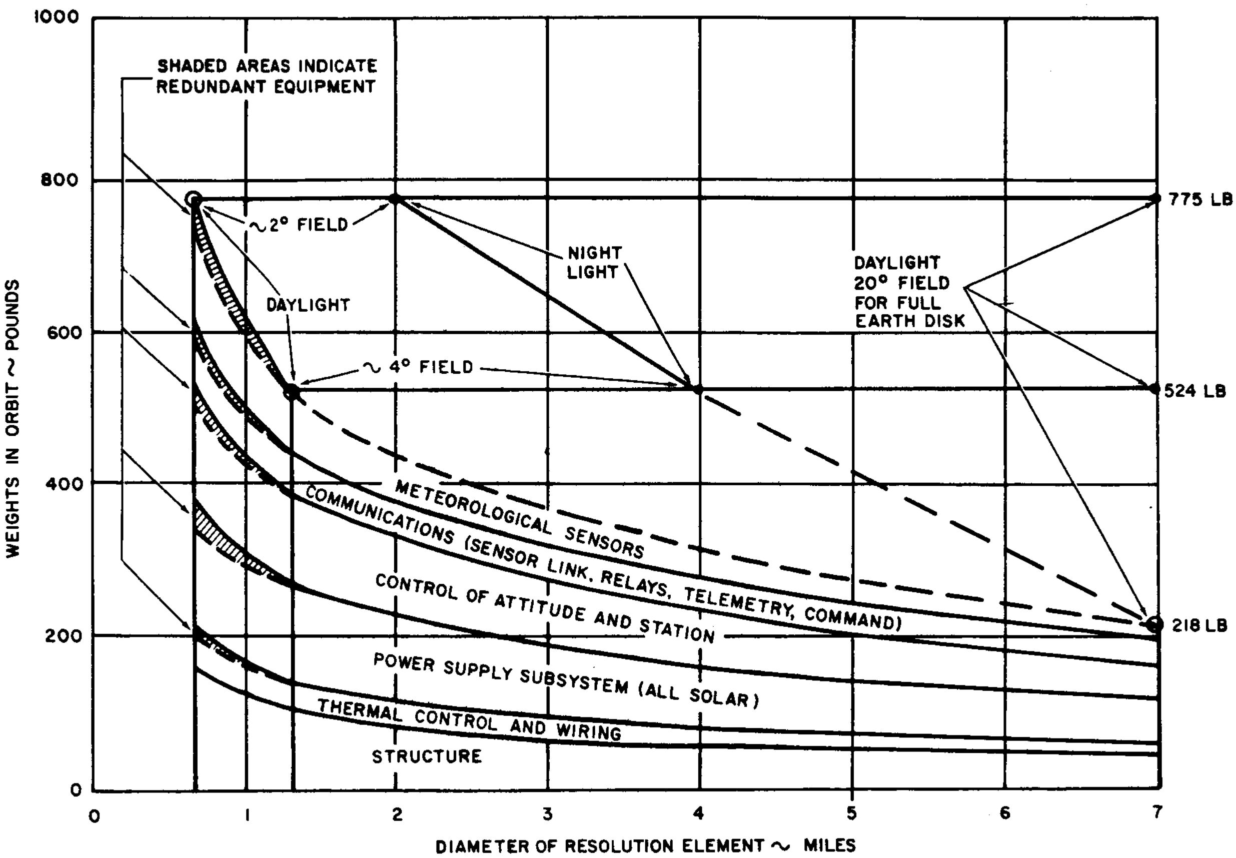



Figure 3-1. Comparative Weights and Resolutions for Three Illustrative SMS Configurations

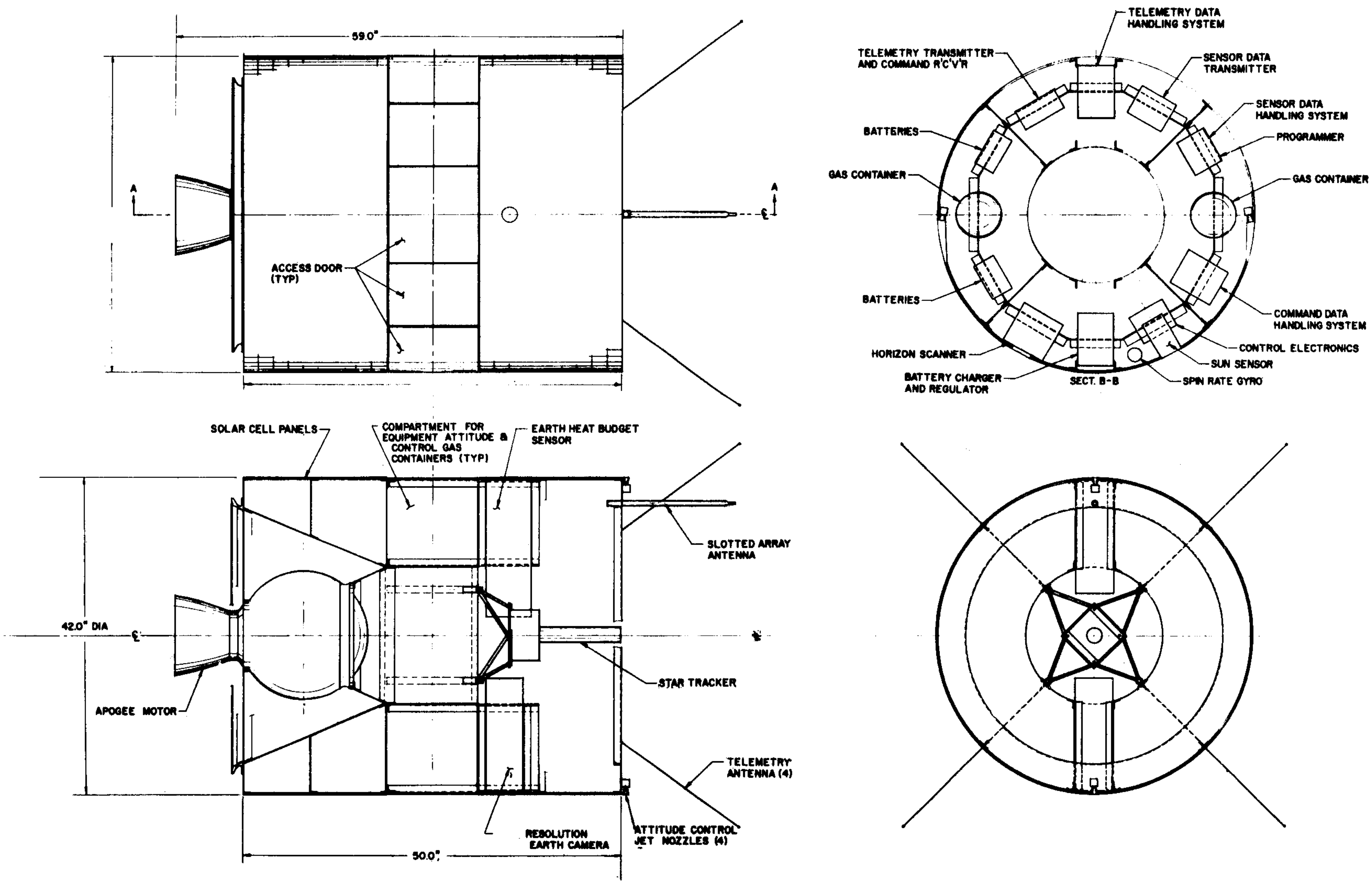



Figure 3-2. Minimum Capability Satellite, Spin Stabilized

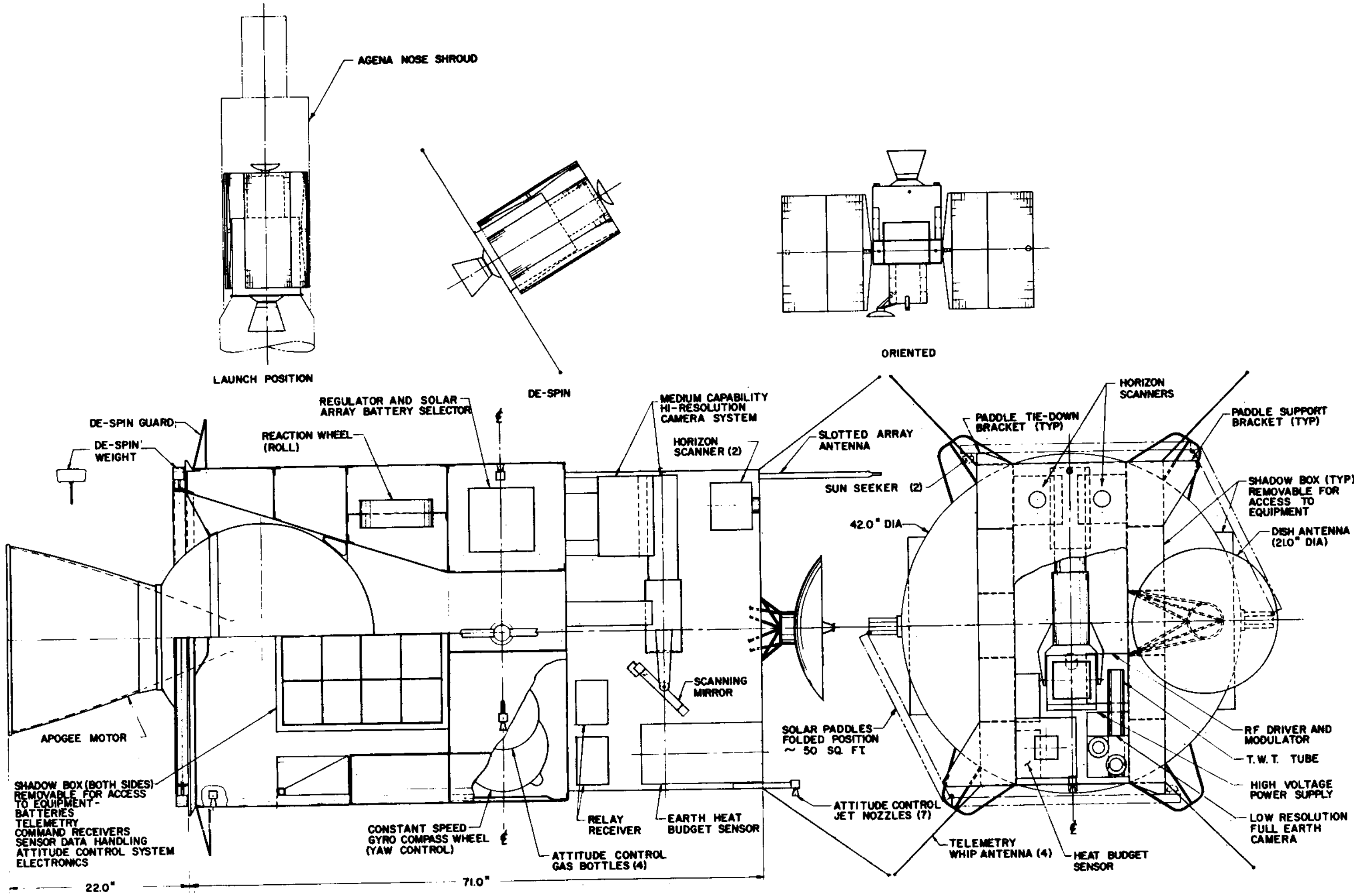



Figure 3-3. Medium Capability Satellite, 3-Axis Stabilized

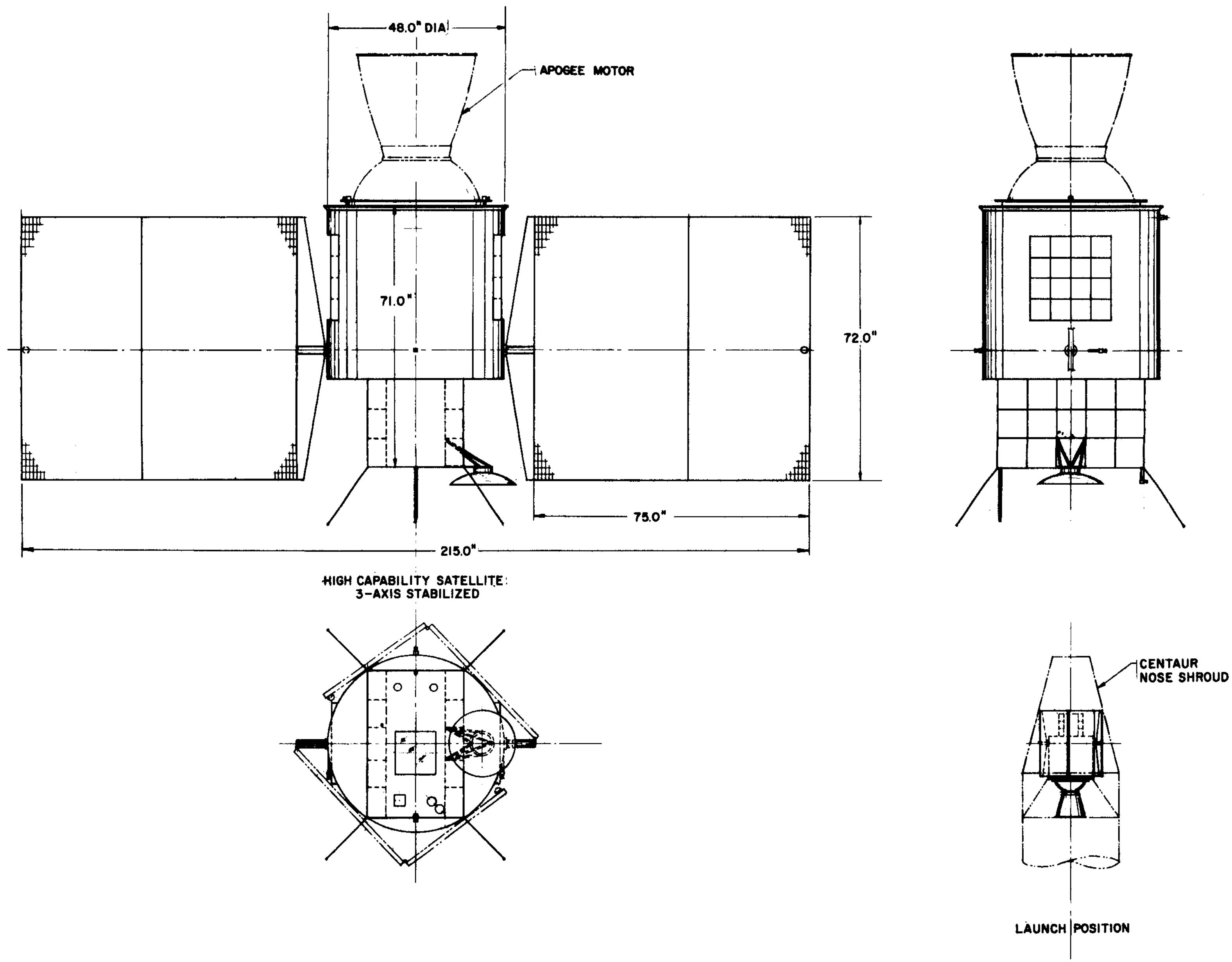



Figure 3-4. High Capability Satellite, 3-Axis Stabilized

TABLE 3-2
PERFORMANCE OF THREE SMS CONFIGURATIONS

| | Configuration Weight (lb) | | |
|---|---|---|---|
| | 775 | 524 | 218 |
| Heat Budget Measurements | | | |
| Wavelength bands (microns) | | | |
| Reflected radiations | 0.2 to 4 | 0.2 to 4 | 0.2 to 4 |
| Emitted radiations | 4 to 40 | 4 to 40 | 4 to 40 |
| Resolution (miles) | 200 | 200 | 8000 |
| Accuracy (degrees Kelvin) | 1 | 1 | 1 |
| Cloud Cover Pictures | | | |
| Earth coverage per frame (miles) | 625x625 | 1250x1250 | - |
| | (plus full Earth disc for all) | | |
| Frames to cover most of disc | 100 | 25 | 1 |
| Resolution (miles per TV line) | | | |
| Daylight areas | 0.65 | 1.3 | 7 |
| For moonless night | 2 | 4 | - |
| For full disc in one frame, daylight | 7 | 7 | 7 |
| Shades of gray ($\sqrt{2}$ basis) | 8 | 8 | 8 |
| Sensor Data Broadcast | | | |
| Frames per minute | 6 | 6 | 6 |
| Maximum baseband frequency (MC) | 0.1 | 0.1 | 0.1 |
| Emitted Power (watts) | 5 | 5 | 2.5 |
| Antenna beamwidth (degrees) | 20 | 20 | pancake |
| Carrier frequency (MC) | ~1800 | ~1800 | ~1800 |
| Telemetry System | | | |
| Information rate (bits/sec) | 210 | 200 | 190 |
| Type of modulation | PCM | PCM | PCM |
| Emitted power (watts) | 2.5 | 2.5 | 2.5 |
| Carrier frequency (MC) | 136 | 136 | 136 |
| Command System | | | |
| Number of commands | 103 | 100 | 85 |
| Type of modulation | PCM and Tone Digital for all | | |
| Tracking Transponder Functions Compatible with NASA Equipment | | | |
| VHF | Yes | Yes | Yes |
| S-Band | Yes | Yes | No |
| Stabilization | | | |
| Sensor Aiming Error (degrees) | 0.5 | 0.5 | - |
| Attitude rate limit-deg/sec | 0.003 | 0.003 | - |
| Station keeping error of latitude or longitude (degrees) | 2 | 2 | 2 |

TABLE 3-2 (Cont'd)

| | Configuration Weight (lb) | | |
|---|---|---|---|
| | 775 | 524 | 218 |
| Power System | | | |
| Average Output (watts) | 288 | 272 | 100 |
| Storage (watt-hours) | 728 | 728 | 330 |
| Thermal Control (°C) | 25±10 | 25±10 | 25±10 |

Relay of processed meteorological data can be accomplished using communication equipment on board. Its characteristics are:

| | | | |
|---|---|---|---|
| Relay to MDA Stations | | | |
| Maximum baseband frequency (MC) | 0.1 | 0.1 | - |
| Emitted Power (watts) | 5 | 5 | - |
| Carrier frequency (MC) | | | |
| Up-link | 2271 | 2271 | - |
| Down-link | 1705 | 1705 | - |
| Relay to Nimbus Stations (modified) | | | |
| Maximum baseband frequency (CPS) | 160 | 160 | 160 |
| Emitted Power (watts) | 2.5 | 2.5 | 2.5 |
| Carrier frequency (MC) | | | |
| Up-link | 148 | 148 | 148 |
| Down-link | 136 | 136 | 136 |

# SECTION 4 - SUBSYSTEM STUDIES

## A. METEOROLOGICAL SENSORS

### 1. General

The principal results of this SMS study program have been presented in preceding sections of this volume, in terms of conclusions and recommendations on the overall synchronous meteorological satellite (SMS) system and program and on specific subsystems. As indicated in the Table of Contents, the present subsection is the first of several which summarize the work on which the conclusions and recommendations are based. These subsections, one on each of the principal subsystem areas, provide a brief review of the work described in greater detail in Volumes 2, 3, 4, 5, and 7 of this report. The principal factors, relationships, limits, and alternatives are covered.

Meteorological sensors constitute the primary payload of the SMS. Resolution levels achievable with synchronous satellites of various weights are indicated in Figure 3-1. All three of the illustrative configurations provide single-frame views of cloud cover over the full disc of the Earth. The two larger configurations also provide higher resolution and day and night coverage for smaller areas. Their optics are aimable, so that the areas can be selected, or if desired, all of the disc can be covered in successive frames.

For the minimum weight satellite (218 lb), a wide-angle heat budget sensor views the whole disc of the Earth and thus gives one integrated value for each observation. For the two larger configurations (524 and 775 lb), the heat budget data is provided by a scanning sensor with a spot size 200 miles in diameter at the nadir.

Figures 4-1, 4-2, 4-3, and 4-4 are schematic diagrams of the four types of sensor subsystems chosen for acquisition of all of the meteorological data just discussed.

### 2. Cloud Cover Sensors

The SMS meteorological sensors will provide 24-hour surveillance of the Earth's cloud cover through the employment of visible spectrum imaging tubes. A very sensitive, miniaturized, Image Orthicon tube camera system, similar in design and characteristics to that being developed by NASA for the Nimbus program, is used, together with catadioptric optics, to obtain high-resolution pictures of cloud cover under both day and night time viewing conditions. A high-sensitivity Vidicon tube camera system provides complementary pictures of the daylight illuminated portions of the full Earth disc, at lower resolution.

The high-resolution Image Orthicon sensor system will be capable of operation over a wide range of illumination values through the use of automatic exposure control by means of apertures, shutters, and filters, and of special tube operating techniques, so that brightness levels of two orders of magnitude may be

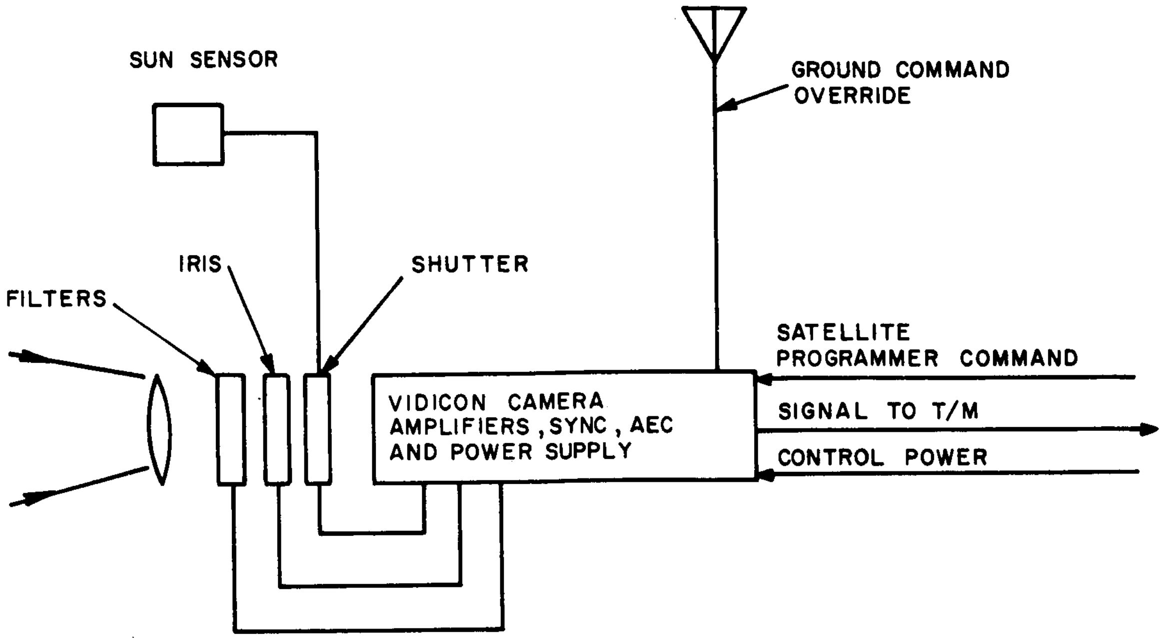



Figure 4-1. Wide Coverage Cloud Cover Sensor

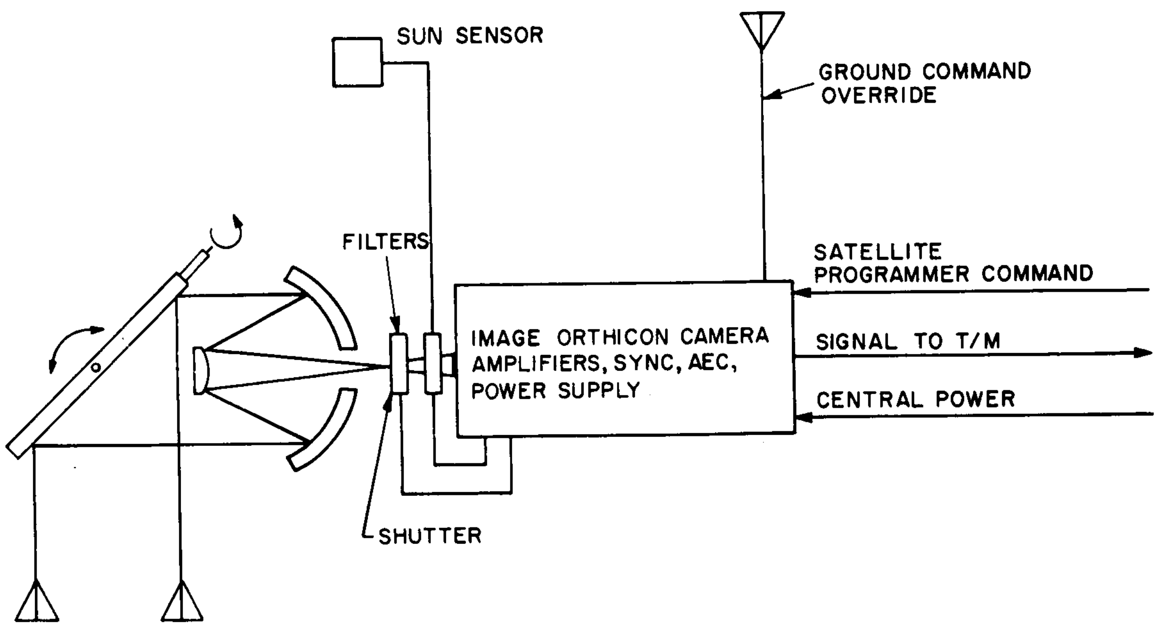



Figure 4-2. Narrow Angle Camera

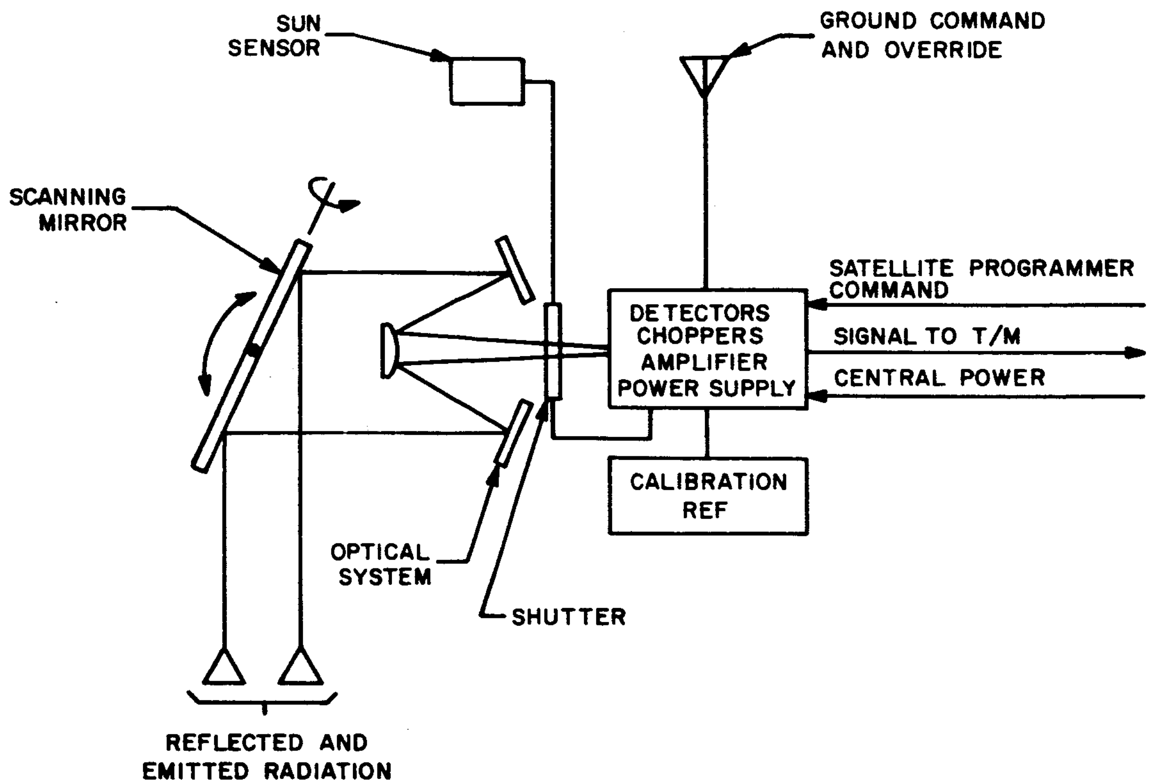



Figure 4-3. Heat Budget Sensor

imaged in a single scene frame. In an SMS of 524 lb, the field of view afforded by the sensor optical systems will provide an Earth area coverage of 1250 by 1250 miles in a single picture. System resolution will be 1.3 miles per TV line at the center of the field for daylight scene brightness conditions from high noon to nominal twilight. When operating at lower scene brightness levels, the resolution capabilities of the sensor hold remarkably well; under conditions approaching starlight, cloud pictures can still be imaged at a resolution of approximately 4 miles. The optical axis of the sensor's lens is directable by means of an aiming mirror so that selected areas of the Earth's surface cloud cover may be brought under surveillance. A high-resolution mosaiking capability for the major portion of the Earth's surface is provided by 25 frames, requiring only about 15 minutes of the reporting cycle.

In an SMS of 775 lb or greater, daylight resolution can be improved to 0.65 mile per TV line, with half the field width; hence, four times the frames are required to cover the same area. Optics of longer focal length are used. For the same night light levels, resolution is improved to 2 miles, and usable pictures are obtainable at lower levels.

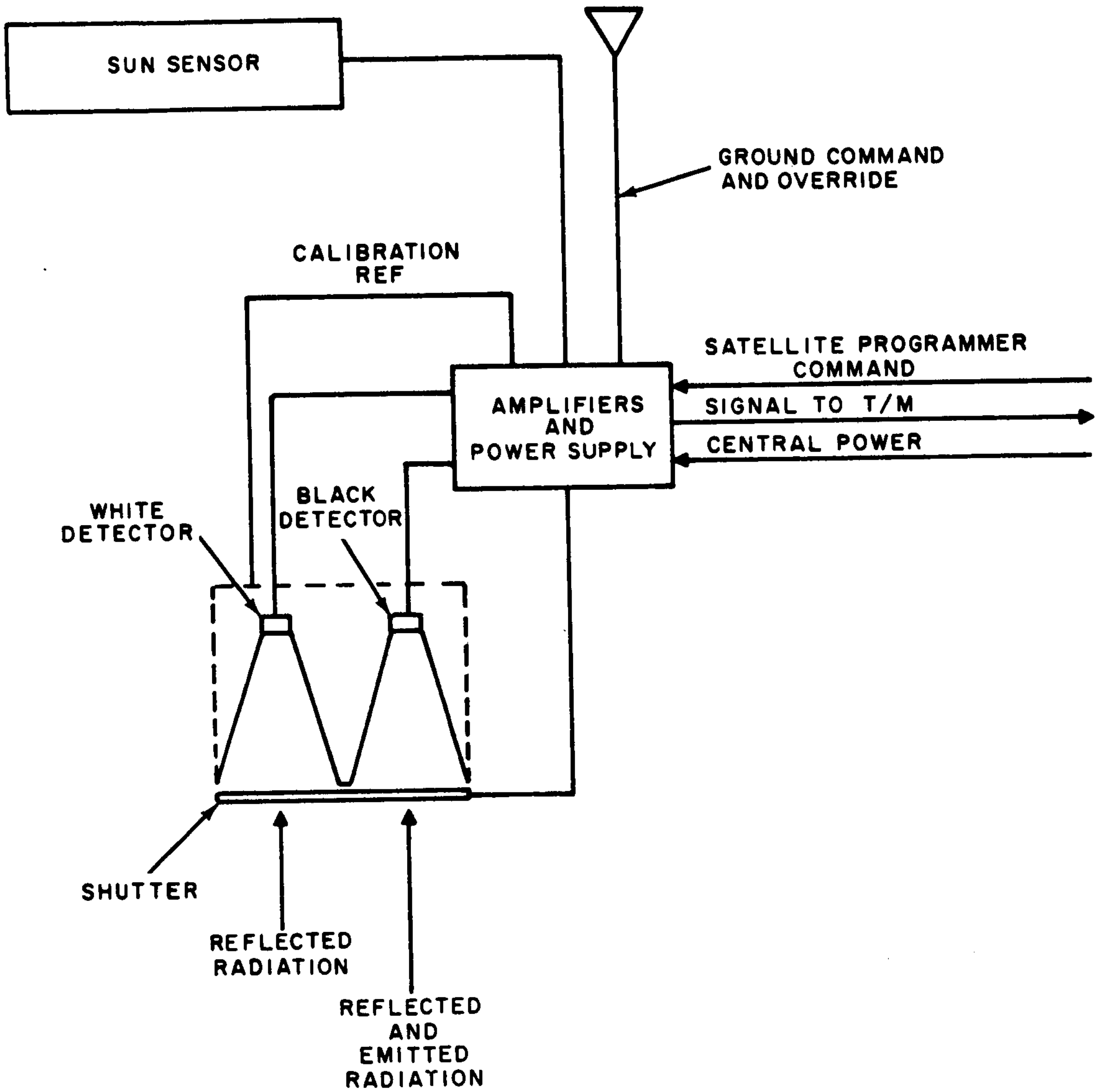



Figure 4-4. Wide Beam Sensor

The Vidicon camera system provides cloud cover surveillance over the entire Earth disc under operating conditions that range from daylight to approximately full Moon illumination. Under these conditions, this sensor provides a nadir resolution of 7 miles for daylight observation of the full Earth disc to approximately 20 miles when the Earth is under the illumination of a full Moon. The camera employs a fixed, short focal length lens whose optical axis is coincident with that of the satellite. In addition to providing wide area coverage of the Earth disc, this sensor may also be employed as a prime reference or monitor for satellite attitude determination and control. Another value of a full Earth disc sensor system is the capability for the ground meteorological observer to see and evaluate a comprehensive view of the weather patterns at a reasonable resolution, and then, having identified areas of interest, he can direct the Image Orthicon camera aiming mirror toward those selected areas to take high-resolution pictures for detailed study.

The accomplishment of 24-hour surveillance of the Earth's cloud cover to fulfill the SMS mission objectives is not without problems. The extremely great range of illumination conditions present (approximately 9 orders of magnitude) are beyond the capabilities of any known single sensor. This problem has been overcome by the employment of two sensors, each capable of performing a majority of the overall mission requirement and capable together of meeting the entire objective.

Among the other problems which are of significance are those of sensor protection, unattended operation and attitude rate control. It is mandatory that the Sun's image be prevented from impinging on the sensitive sensor imaging surface. Precautions, such as sensor shutdown and/or employment of occultation shutters, will prevent catastrophic or partial damaging of the sensor due to accidental exposure to the Sun. Figure 4-5 shows how little observation time would be lost if sensor operation is suspended when the Sun is near the edge of the Earth.

Unattended operation of the SMS sensory systems during a one year orbital life is a formidable requirement. Unlike the Tiros and Nimbus satellites, the SMS can be in constant contact with the ground control station. This unique advantage of a synchronous satellite can be exploited by the insertion of a ground observer and ground control instrumentation into the camera control loop to enhance the attainment of a reliable one year orbital life. Other advantages gained are rapid response to data deficiencies or special situations.

Satellite attitude angular rates can impose limits on the maximum exposure time usable without image smear beyond one half of a TV line. Hence these rates limit both resolution and the lowest light levels at which useful pictures can be obtained. Values quoted in this report on these items are those for pitch and roll rates limited to 0.003 degree per second by the control systems selected for the satellite configurations. Closer limits on these rates would improve the sensor performance by permitting longer exposure times. Subsystems for achieving and bettering these rate limits are reviewed briefly in Section 4.D of this volume and at more length in Volume 4.

The performance levels quoted in this report for visible light cloud sensors are believed to be conservative. Of necessity, they were derived from available data on imaging tube characteristics. All of these data are determined for tube operating conditions corresponding to typical TV applications, with scanning rates of 30 frames per second. Data is not available as to how much the characteristics change with the longer exposure times and scan times (for the latter, about 300 times longer) of SMS applications. It is generally believed that substantial improvements are to be expected in low light sensitivity and in resolution, on the latter by factors such as 3 or 4. For performance estimates in this report, a factor of 1.5 was assumed.

The NASA Universal Camera Tube Tester now under development will provide the means for replacing these assumed factors by measured values directly applicable to SMS applications.

With firm data on tube characteristics and on weight limits imposed by available boosters, it will be appropriate to re-examine the design of the sensor optical system for opportunities to save weight or, alternately, to utilize a weight margin for improved performance.

Volume 3 presents the results of comparisons among a wide variety of additional cloud cover sensors

### 3. Sensors for Heat Budget Data

The Earth's heat budget involves both reflected solar and emitted terrestrial radiation. The solar radiation of interest for heat budget purposes includes the ultraviolet, visible, and infrared radiations having wavelengths between 0.2 and 4.0 microns. The terrestrial emissions consist of infrared energy at wavelengths longer than 4.0 microns radiated by the Earth and its atmosphere. The SMS heat budget measurement considers radiant energies from 0.2 to 40 microns which encompass better than 90 percent of the total spectral radiant energy.

The major portion of the energy loss due to reflection occurs at short wavelengths (0.2 to 4.0 microns) while the major portion of emitted energy loss takes place in the longer wavelength region (4.0 to 40 microns). Heat budget data are acquired by measuring and comparing the radiation losses in at least these two spectral bands.

The absorption characteristics of the Earth's atmosphere for various spectral bands are related to the concentration of the absorbing medium. In general, the dominant absorbing component of the atmosphere is water vapor ($H_2O$). Although water vapor comprises less than 3 percent of the atmospheric gases, it accounts for nearly all of the gaseous absorption of terrestrial emissions. Smaller, but noticeable additional absorptions, are due to carbon dioxide ($CO_2$) and ozone ($0_3$), with the absorption by carbon dioxide at least one order of magnitude greater than that of ozone.

In atmospheric window regions (spectral bands of relatively high transmittance lying between two regions of low transmittance), the Earth radiates energy to space at a rate associated with its representative black body temperature of 288°K. In absorption bands, the escaping radiation comes from the stratosphere with a representative temperature of 218°K. The SMS study considered

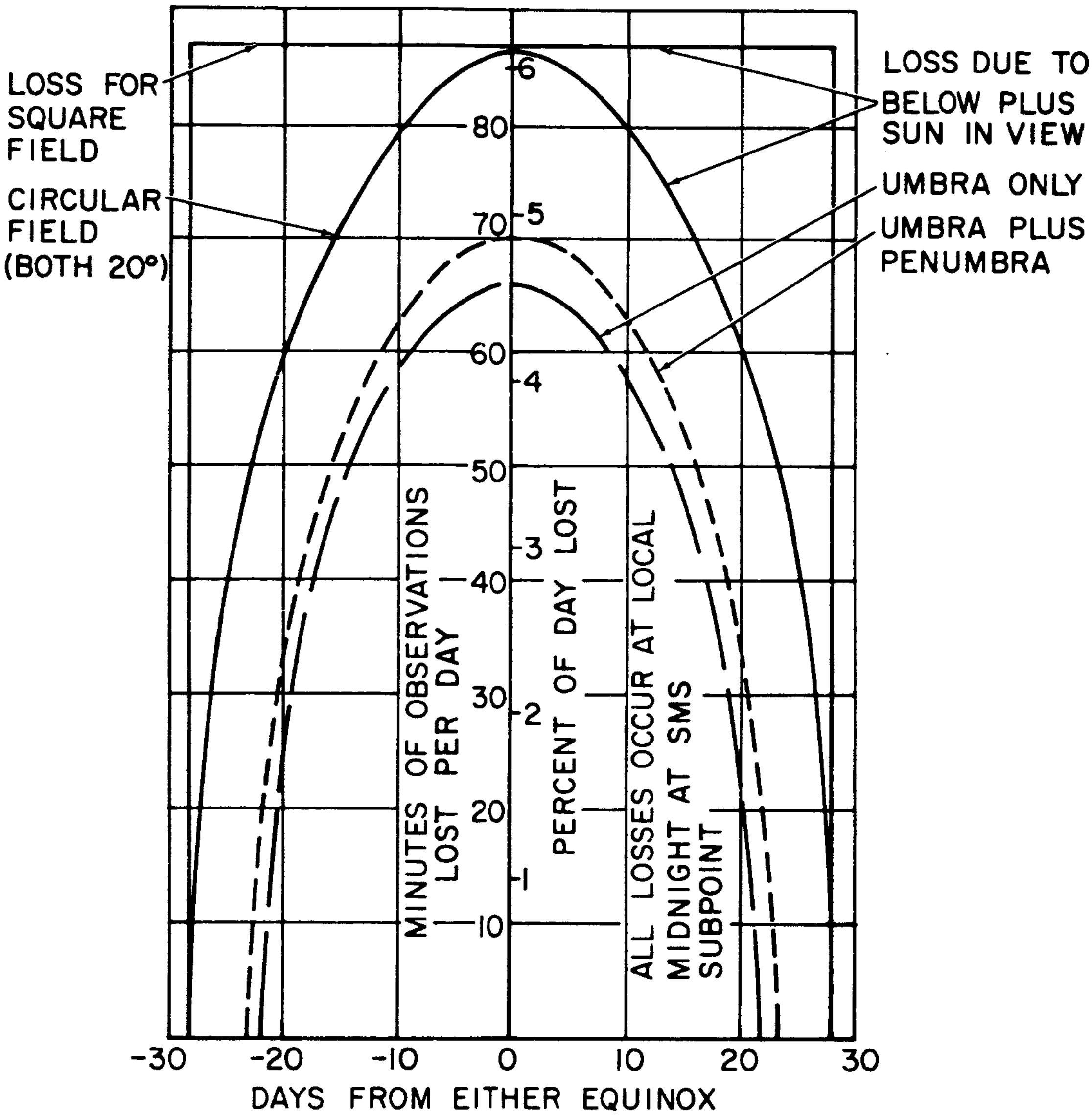



Figure 4-5. Losses of SMS Observation Time Due to Sun and Shadow

black body temperatures over the broader range of from 100 to 400°K. Figure 4-6 shows a curve illustrating the spectral radiance from 288 and 218°K black bodies together with the resultant radiation spectrum.

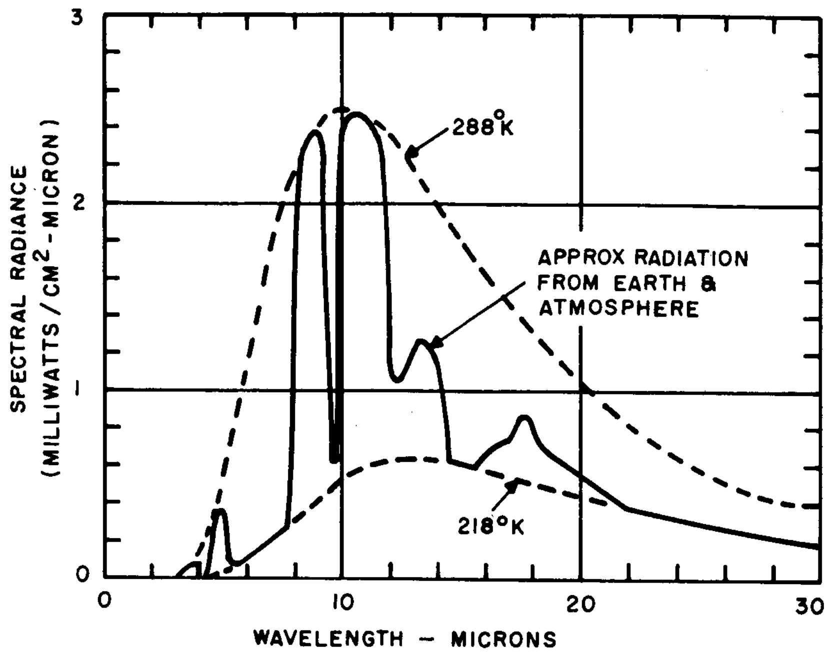



Figure 4-6. Typical Terrestrial Radiation Spectrum

Cloud cover is a dominant factor in both solar and terrestrial radiation considerations. Clouds must be considered black body radiators because of their great absorption capabilities.

Clouds absorb and re-emit radiation in all of the infrared wavelengths including the water vapor windows. Normally, their temperatures decrease with altitude. The cloud layer absorbs radiation from below and re-emits radiation both up and down. The result, as far as radiation to space and the SMS is concerned, is the creation of a radiation surface as effective as the ground but at a considerably lower temperature and, hence, longer wavelength.

Estimates made by various investigators show that the terrestrial emission levels vary as a function of latitude while variations between daytime and night time values can sometimes be ignored.

Conclusions drawn as the result of the investigations for the SMS program show that terrestrial radiation is entirely dependent upon the transparency of the atmosphere, which varies principally with the amount of water vapor, carbon dioxide, and cloud cover.

The instrumentation for the measurement of the Earth's heat budget requires the employment of radiation detectors which are sensitive in the spectral regions of from 0.2 to 40 microns. Thermistor bolometers offer the best assurance for a practical, reliable, and realizable measuring instrument for the SMS mission. The thermistor has consistently been selected, over more sophisticated detectors, as the most efficient thermal detector for spaceborne infrared detection.

Heat budget instrumentation can be as simple as a modified V. E. Suomi type instrument, where the radiation energy is impinged on separate black and white thermistors located at the apexes of geometric cones, or more complex but higher resolution scanning instruments which utilize reflective optics for gain. In the latter systems, the total energy content of the instantaneous area viewed by the optics would be divided, so that the reflected radiation would impinge on one detector while the emitted energy would impinge on a second detector.

Volume 3 describes and compares several types of possible heat budget sensors.

4. Problems of Illumination and Dynamic Range

All meteorological sensors considered for the SMS are of passive nature, dependent on infrared emissions and reflected radiation from clouds and Earth. Sources of reflected radiation are natural sources such as the Sun, the Moon, and the stars. Emitted radiation is filtered once by the atmosphere before reaching the SMS sensors. Solar radiation is filtered by the atmosphere once before reflection and once again after it. In both cases, the degree of filtering varies with the path length and with composition of the atmosphere along the path.

Basic to selection, design, and evaluation of the meteorological sensors is some understanding of the radiation energy levels, contrast, and dynamic range characteristics within fields of view.

5. Radiation Levels and Problems

For the purposes of the SMS study, a homogeneous model of the Earth's atmosphere was developed. This model consists of the Earth surrounded by a homogeneous atmosphere of nearly 8 kilometers thickness. The various constituents of the visible spectrum relating to cloud cover imaging were identified and analyzed, to permit prediction of levels of illumination to be incident upon the meteorological sensors. The solar radiation reflected to the satellite varies as a function of object altitude, solar altitude, albedo, etc. In addition, contributions of radiation from other sources such as moonlight, airglow, etc., become important, since, during the majority of time, some night scenes will be within the satellite field of view.

Extremely wide variations in illumination will be encountered by a meteorological satellite viewing the Earth from its stationary vantage point. Consider the situation for example, when it is high noon at the subsatellite point. As seen from the satellite, the range of illumination on the Earth disc includes sunrise on its west edge, sunset on the east edge, and high noon maximum in the center. In order to illustrate, in a practical manner, several situations have been postulated and are depicted by Figure 4-7. For this figure, two Sun positions were selected and two atmospheric conditions were assumed, one with no clouds, and the other with complete high albedo cloud cover at 46,000 ft. In the SMS field of view, the Earth's surface along the equator consists of water (Atlantic Ocean) from 10 to 50°W longitude, land (northern section of South America) from 50 to 80°W longitude, and again water (Pacific Ocean) from 80 to 170°W longitude. Examination of these plots shows that the maximum irradiation to be expected on the photocathode of the meteorological sensor for an f/0.5 lens and a cloud at an altitude of 14 Km, having an albedo of 0.7, is about 8 x $10^3$ foot-candles when it is noon at the satellite subpoint, and 5 x $10^3$ foot-candles morning or evening there. For night time illumination, the incident irradiation on the sensor could be lower than $10^{-6}$ foot-candles. This spread corresponds to a dynamic range of more than 9 orders of magnitude.

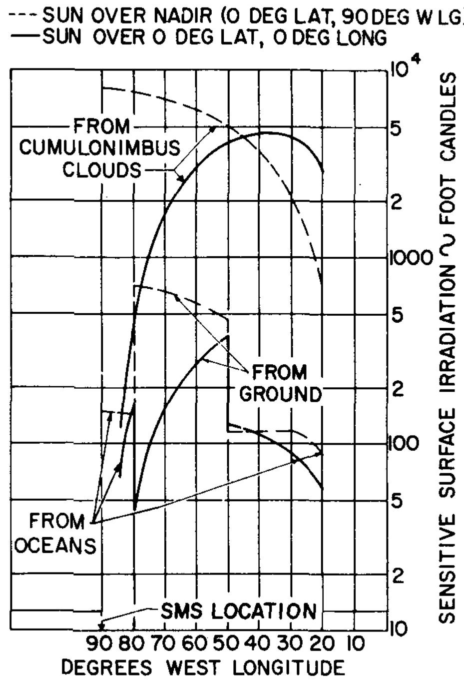



Figure 4-7. Variation of Light Level with Longitude and Sun Angle

The fundamental problem associated with a 9 order dynamic illumination range is that, to date, no single sensor has been devised which can accommodate, in a single scene, such a large variation of illumination. Use of special camera tube operating techniques and attenuating filters can increase a basic 35 to 1 sensor tube dynamic range to perhaps 2 orders of magnitude which falls short of the range requirements. If a single frame picture of the full Earth disc is required, the best results would be obtainable with adjustment based on the daylight areas. For complete coverage, a more practical approach is to employ a mosaic of pictures in which the range of illumination within any on picture is limited by use of a narrower field. A full Earth composite of the individual pictures, recorded with different adjustments, would then fulfill the requirement, with broader coverage and higher quality.

Since the illumination levels are predictable, on a daily and hourly basis, it does not seem unreasonable to plan for programmed surveillance along lines of relatively constant illumination to provide greater assurance of maintaining the sensor within bounds of its dynamic operating range.

### 6. Conclusions and Recommendations for Meteorological Sensors

(1) Sensors for the SMS mission can be developed with present technology.

(2) The Vidicon is the best available sensor tube for imaging cloud cover over the whole Earth disc in one picture.

(3) The Image Orthicon is recommended as the best sensor for narrow angle pictures. Some development work will be required to perfect a flight article capable of unattended operation for one year in orbit.

(4) Standardized measurement of tube characteristics, as by the NASA Universal Camera Tube Tester, will be of considerable assistance in selecting sensor tubes. Data now available is mostly derived from tests related to TV studio use rather than for spacecraft applications.

(5) In the development of SMS sensors for future use, the Image Dissector should be considered. This sensor, at present, suffers from poor sensitivity. Its inherently wide dynamic range and high resolution come closer toward meeting the full Earth disc imaging problem requirements than any other sensor. The low sensitivity can be overcome by use of longer exposure times.

(6) Future development of tubes should stress the obtaining of more lines per tube in order to obtain greater resolution over the same area without resorting to larger optics.

## B. METEOROLOGICAL COMMUNICATIONS

### 1. General

Accurate delivery of meteorological data is second in importance only to the sensing function. The utility of the communication system can be increased by using the equipment for other functions. By appropriate choice of frequencies, the satellite sensor data transmitter can also be utilized as part of the system for relay of processed meteorological data from the control and data acquisition (CDA) station to other ground stations. The satellite relay equipment can also serve as a tracking transponder compatible with the Goddard Range and Range Rate equipment. It is also possible to relay data information to Nimbus APT stations, properly modified, by using the command receiver and telemetry transmitter as relay equipment.

The SMS sensor communication links provide the following services:

(1) Delivery of meteorological data to the CDA station with a large safety margin for fade and aging degradation. Data is also transmitted to multiple data acquisition (MDA) stations with a lower margin.

(2) Wideband relay of processed meteorological data from the CDA station to MDA stations with little additional complexity in the spacecraft.

### 2. Sensor Data Link

#### a. Objectives

The objective for the sensor data transmitter is to provide a reliable channel for the transmission of both cloud cover and heat budget data from the SMS vehicle to the control and data acquisition (CDA) station and to many other stations within view of the satellite. Therefore, the antenna beam pattern must cover the full Earth disc. The CDA station can have a large, high-gain antenna for this communication link, but the multiple data acquisition stations (MDA) are not likely to be able to support as large an antenna facility. Therefore, some sacrifice of safety margin may be expected for them.

This study examined the critical design parameters relating to the link and illustrated their application to specific satellite configurations. These parameters, their relationships, and a table of system design data for a number of configurations are summarized in the following paragraphs. The general conclusion is that satellite transmitters and power supplies can be selected or designed, within the present state of the communications art, to meet the requirements for data quality, quantity and frequency of reporting. For a more detailed analysis, refer to Section 1 of Volume 5.

b. Principal Design Parameters

A careful examination of the design parameters for this communications link for the SMS showed that many of them are firmly established by the ground rules of the problem and cannot be left to the discretion of the designer.

Specifically, the choice of carrier frequency is, to a large extent, dictated by authorized spectrum allocation. Free space attenuation and antenna gain increases with frequency, but antenna noise temperature decreases with frequency (up to about 8 KMC). Estimated channel bandwidth requirements for slow scan TV sensors (0.1 to 1.0 MC) suggest the selection of the frequency spectrum in the 1700 to 2200 MC band. For the chosen frequency of 1800 MC, the free space losses, between the SMS and a ground station at the sub-point, total 189 DB. For receiving stations located at any other point, slight additional losses are incurred, but in most cases these will not exceed 1 DB.

The antenna requirements for both the ground and space terminals are also largely determined by the system operational characteristics. For an Earth oriented, 3-axis stabilized SMS, a directional antenna can be tolerated, but the maximum gain is limited by the desired requirement that full Earth coverage be provided in order to enable other (MDA) stations to receive the same data. Since the Earth subtends a 17-degree angle at synchronous altitude, an antenna beam of 21 degrees was chosen to allow a safety factor for small stabilization errors. Since the gain of an antenna is related directly to its beam pattern, the antenna gain for the SMS vehicle will be 18 DB (regardless of frequency). A spin-stabilized satellite cannot utilize a simple directional dish antenna, so either of two approaches can be taken; a multi-element beam switching configuration that would approach the directivity of a dish at the expense of a heavy and complex switching system; or a slotted array with a "pancake" pattern that provides gain in the plane perpendicular to the spin axis without requiring a switching system. The latter method, in spite of its relatively low gain (about 6 DB), was chosen for the light weight spin-stabilized configuration because of the resultant weight saving.

The ground antenna at the CDA station was treated as a parameter, but the upper limit of gain was determined by the maximum dish diameter of 85-ft, a limit established by NASA. At 1800 MC, this antenna would exhibit a gain of 51 DB. At the same frequency, a 28-ft diameter dish (a likely size for MDA stations), would provide a gain of 41 DB.

In addition to parametric limitations imposed by the operating conditions, certain other parameters are limited by the availability of components. In particular, these are the transmitter power efficiency and receiver noise temperature. Four transmitter systems were considered for this application. Three, the travelling wave tube (TWT), the amplitron, and the voltage-tuned magnetron, can be used as final stage power amplifiers; one, the all-solid state unit, is a varacter diode chain that can be used as driver and final output stages. The relative advantages of these components are discussed in Volume 5, with the conclusion that the TWT offers the best choice of power gain, wide frequency band, and long life in spite of its overall efficiency of about 20 percent, which is not outstanding compared with the amplitron's capability of better than 50 percent.

The ultimate sensitivity of any receiver is largely determined by its effective noise temperature. In the frequency band chosen, the contribution from the antenna is less than 20°K, so that the receiver front end noise temperature is likely to be the dominant factor. Three types of low noise front ends were considered. In order of decreasing noise temperature, they are tunnel diode, parametric, and maser amplifiers. The tunnel diode amplifier is small and inexpensive, but produces a noise temperature in excess of 300°K.

The noise temperature of a maser (when properly cooled) can be held to nearly 10°K. This device is bulky and expensive. The parametric amplifier selected for this application while not compact, is less costly and would not impose a burden in a ground station application. Its noise temperature of 150°K (or a noise figure of 1.8 DB) results in a conservative receiver design.

The remaining critical parameters in a communication link design relate to the character and quality of the data being transmitted, and the extent of degradation that can be tolerated.

For purposes of comparison, a maximum data rate of 200,000 picture elements per second was assumed for each of the sensor outputs, and no two of the major sensors would be operated simultaneously. Furthermore, the peak signal to noise ratio is judged to be less than 40 DB. (For justification of these values refer to Volume 3 of this report.)

In order to minimize the resultant link degradation on the sensor video, a 46 DB output signal to noise ratio was established as a requirement. The 6 DB improvement over the input data quality was expected to produce no more than a 1 DB overall system degradation in the sensor output. In order to preserve a good waveform, a video base bandwidth of 100 KC was preserved throughout the transmission system.

Finally, an estimate must be made for incidental line losses, polarization errors, component degradation, and unpredictable fades. In spite of all these losses, an adequate carrier to noise (C/N) power ratio must be obtained in the receiver so as to exceed the modulation threshold. This threshold value varies with the particular modulation methods employed.

Two modulation techniques were examined in detail: analog FM with feedback (FMFB), and digital bi-phase PCM. For the particular channel requirements postulated above, the threshold C/N ratios are 17 DB and 12 DB, respectively.

c. Illustrative Examples

Sample calculations based upon the two methods of modulation just discussed are presented in the study, and the results are summarized in the following tabulation.

## Summary of Sensor Data Link Design Parameters

| | | |
|---|---|---|
| Carrier frequency | 1800 MC | |
| Fixed Gains and losses | Gains | Losses |
| Free space atmospheric attenuation | - | 189.5 DB |
| SMS antenna | 18 DB | - |
| Ground antenna | 51 DB | - |
| Margin for incidental losses | - | 19 DB |
| Total net losses | - | 139.5 DB |
| Noise temperature | | |
| Antenna noise temperature | 28°K | |
| Receiver noise temperature | 150°K | |
| Total effective noise temperature | 178°K | |
| Video base bandwidth | 100 KC | |
| Desired minimum receiver output S/N | 46 DB | |
| Comparison of Modulation methods | FMFB(analog) | PCM(digital) |
| Modulation characteristic | Modulation index = 16.3 | Code = 6 bits |
| Required IF bandwidth | 0.52 MC | 2.4 MC |
| Receiver noise power | -149.3 DBW | -142.5 DBW |
| Threshold C/N | 17 DB | 12 DB |
| Minimum receiver input power | -132.3 DBW | -130.5 DBW |
| Total net gains and losses | 139.5 DB | 139.5 DB |
| Required transmitter power | 7.2 DBW | 9.0 DBW |
| or | 5 W | 8 W |

The comparison based upon the above illustrated modulation techniques shows that in spite of a number of conservative design assumptions, the total power requirements at the transmitter are quite reasonable, and even with the 20 percent efficiency of the TWT, would not impose much more than a 30-watt load on the satellite's power system. The power ratio (less than 2 DB) between the analog and digital modulation methods is small compared to the margin of error in the estimates upon which the above illustrations are based, hence the two methods can therefore be considered nearly comparable. Should a S/N much greater than 46 DB be required of the system, the digital method might show a greater advantage; but should the converse be true ( a more likely case) the analog (FMFB) method would be a better choice.

For the above, gain reductions due to a spin-stabilized satellite antenna pattern (12 DB less than the directional antenna) or a 28-ft ground antenna (10 DB less than the 85-ft antenna) would be applied against the 19 DB of incidental losses without changing the other parameters. The net effect would be to reduce

(2) Spacecraft to Ground Link (Down-link)

Transmitter (Space-SMS vehicle)

| | |
|---|---|
| Frequency | 1705 MC |
| Power | 5 W (average) |
| Modulation index (FM) | 2 |
| Antenna diameter | 21 in. |

Receiver (Ground-MDA stations)

| | |
|---|---|
| Noise figure | 3.5 DB |
| Antenna diameter | 30 ft |
| Received C/N | 11 DB |
| Safety Margin (incidental losses) | 8 DB |
| Min. output S/N | 22 DB |

Examination of the above figures illustrates the requirements for the ground transmitting station (CDA) and the remote receiving stations (MDA). The CDA station requires an 85-ft antenna for the receiver in sensor data link, and the same dish with a separate feed can be used for the relay transmitter. The receiver characteristics required for the MDA indicate a modest 30-ft fixed antenna and relatively simple (uncooled) tunnel diode preamplifier with a noise figure of 3.5 DB. The spacecraft receiver noise figure of 5 DB is even less stringent. In general, the above design should not impose unrealistic requirements upon the capabilities of existing components.

The channel capacity of 100 KC is quite broad and can be used to send many simultaneous facsimile transmissions of the type discussed previously. The minimum S/N of 22 DB is more than adequate for black and white (line) reproductions and should be capable of handling at least seven shades of gray.

The tabulation above is for the case of simultaneous delivery of sensed and processed meteorological data. The tubes and power systems included in all three illustrative configurations are adequate for this.

Several advantages are seen in making these two transmissions sequential rather than simultaneous. First, the MDA stations could operate with one receiver-recorder channel instead of two, with a reduction in equipment cost. Second, the spacecraft transmitter would be shifted off of the frequency of the range and range rate tracking system for all relay transmissions, thus eliminating possible interference with tracking of other satellites. Third, power requirements would be reduced, providing either a weight saving or a safety margin.

Relay of processed meteorological data to Nimbus APT Ground Station equipment presents quite different problems. The characteristics of the ground receivers and recorders are already established for the Nimbus system.

To deliver processed data of bandwidth comparable to that received from Nimbus would require impossible power levels and/or antenna gain in the SMS. However, it is possible to narrow the bandwidth and raise the frequency so as to deliver in facsimile form data such as sections of currently used hemispheric weather analysis charts.

The Nimbus APT receiver could be changed to use a bandwidth of 1.5 KC in lieu of the present 20 KC bandwidth with a gain in signal to noise ratio of 12 DB. Because of the narrower bandwidth, the resolution of the lowered base-band frequency system would be approximately 1/10 that of the original system. The lower resolution could be compensated for by sending a given picture in sections so that any desired resolution could be received. This, in effect, would be equivalent to increasing the transmission time. This method would not require a change in the APT antenna or facsimile recorder. Another possible modification would be to change the speed of the facsimile machine by changing gear ratios so that a longer readout time would be available. The loss due to the increased distance to SMS could be partially made up by replacing the present antenna with one of higher gain.

All of these possibilities are predicated on use of the telemetry and command equipment (described in the next subsection) to perform the functions of a relay transponder, in preference to adding special relay equipment in the SMS. Volume 5 provides detail analysis on the problems of relay to Nimbus equipment.

### 4. Conclusions and Recommendations for Meteorological Communication Equipment

(1) Meteorological data can be transmitted from SMS to the ground by equipment which can be designed and built at the present time.

(2) The usefulness of the sensor data link can be increased by using it as part of the transponder for the Goddard Range and Range Rate System and also as part of a system to relay processed meteorological data from the CDA station to MDA stations.

(3) Some modification of the Nimbus APT stations would permit them to receive processed meteorological data through an SMS repeater.

## C. OPERATING COMMUNICATIONS

### 1. General

Operating communications include all of the communications functions not immediately involved in delivery or relay of meteorological data. Hence, the term includes telemetry, command links, and the functions of tracking SMS position and velocity, both during its ascent and injection into orbit and for station keeping purposes thereafter.

Telemetry is considered needed to provide an adequate basis for commands. It must provide data on many operating conditions in the satellite. Also, it is required for verification of commands as received in the SMS prior to their execution.

There is obvious need for one or more command links to control the ascent, to modify satellite operating conditions, and to actuate or change programs of meteorological observation and reporting.

During the phases of ascent to and injection into orbit, there is a need for tracking data as to satellite position, velocity and acceleration.

None of the equipment for operational communications involves development problems.

### 2. Command Link

The command link provides a means for communicating with the satellite from the ground control station. As with all command systems, reliability is of major importance. Hence, verification techniques are desirable. Command verification involves storing the command data in the SMS while transmitting it back to the ground over the telemetry link. After verification, a return signal over the command link is sent as an "execute" command.

Estimates made during the study of the total number of commands have reached the equivalent of 100 "on-off" items; but this figure is very sensitive to system operational procedures. If additional continuous adjustments of the sensors are required, the number of commands would increase significantly. Nevertheless, for purposes of this study it was assumed that the command link requirements for the SMS do not differ significantly from those of other current spacecraft, such as OGO; hence, standard formats and standard equipment are applicable.

Although two basic formats are recommended by NASA (i.e., tone digital or PCM) the estimated number of commands exceeds the capabilities of the tone digital code, so that the PCM format appears more attractive. During the launch and ascent phase, only a few commands will be needed, so the tone digital format might be adequate during this period. Because most global communications centers are being fitted with tone digital command units, the combination of tone digital during launch ascent and injection and PCM during normal on-station operation would appear to satisfy most requirements.

The command link proposed for the medium capability SMS consists of a 5000 W transmitter radiating at 148 MC by means of a 20 db 3 bay helical antenna. At the receiving end, the command receiver shares a turnstile antenna formed of quarter wave whips with the 136 MC telemetry transmitter. The signal is received (after appropriate isolation) in an AM superheteorodyne single conversion receiver. This yields a 25 db signal to noise ratio. The output of the receiver is then decoded and stored until the command has been properly executed. Reliability requirements dictate that two receivers with independent power supplies be used in parallel.

3. Telemetry Link

The principal function of the telemetry link is to transmit housekeeping data and command signal verification from the SMS to the ground control station. Telemetry is required for monitoring the performance of the various satellite subsystems during all phases of its operation. These subsystems include the following:

(1) Meteorological sensors

(2) Attitude control

(3) Station keeping

(4) Communication subsystem

(5) Power supply subsystem

(6) Thermal subsystem

(7) Launch performance

(8) Command signal verification

(9) Transponder for meteorological data relay

(10) Other satellite instrumentation

The telemetry link can be divided into two parts. The first is the conversion and processing of the information obtained from each of the subsystems. The second is the generation, transmission and reception of RF energy. These parts are interrelated and must be considered simultaneously, to arrive at an optimum telemetry subsystem.

Each item of telemetry data will have characteristics which determine the information rate and, hence, the bandwidth requirements. These characteristics include accuracy required, rate of change, sampling frequency, duty period and redundancy required. The bandwidth of the telemetry channel affects the telemetry power required.

The results of the study indicated that approximately 100 points on the 524 lb SMS would be monitored. PCM appears to be the most suitable technique. The required information capacity would be a maximum of 200 bits/sec for the medium capability SMS. The data processing portion of the telemetry system would consist of signal conditioners, analog to digital converters, and a multiplexer. The resultant pulse code modulation would frequency-modulate the telemetry transmitter.

The telemetry transmitter for this SMS would be a solid state unit operating at 136 MC with an output of 2.5 W. It would feed a turnstile omnidirectional antenna consisting of four quarter wave whips. A duplexer system would permit both telemetry transmission and command reception with the same antenna. The ground equipment to receive the telemetry information could consist of equipment already in use for other satellites.

4. Satellite Beacon and Tracking

A beacon and tracking system for the SMS is required for the following purposes:

(1) For timing and guiding the orbital injection operations

(2) For detecting and measuring later drifts of the orbit beyond acceptable limits during station keeping

(3) For obtaining reference coordinates of cloud pictures and other meteorological data

To perform the tracking function, special instrumentation such as a frequency coherent transponder will be required aboard the SMS. To conserve weight, it is desirable that this instrumentation use as much of the existing equipment already on board.

The tracking requirements for the SMS will vary with each phase of operation. During the launch and parking orbit phase, it is desirable to determine the orbital parameters such as period, eccentricity and inclination. During the transfer ellipse, it is desirable to measure the trajectory parameters such as position and velocity. During station keeping, angular position alone may be adequate. Each of these phases may, therefore, require a different method of tracking.

When the SMS is launched and being placed into the parking orbit, a 136 MC beacon would be required by Minitrack stations. This would consist of setting the telemetry transmitter in a beacon mode by removing modulation and reducing power output to 0.25 W (to conserve power). Minitrack stations at various points along the parking orbit can track the SMS and obtain the orbital parameters. In addition, the Goddard Range and Range Rate (RARR) Stations can acquire the SMS by means of this VHF beacon.

During the transfer ellipse phase, it is necessary to measure position and velocity of the SMS. To perform these measurements and for observing station corrections, it is proposed to use the Goddard Range and Range Rate system.

For the medium and high capability SMS, an S-band coherent transponder is proposed for RARR measurements. This would consist of the S-band (1705 MC) sensor data transmitter output stage in conjunction with an S-band (2271 MC) receiver and linear translator. The maximum sidetone frequency for ranging would be 100 KC. The transponder system could also be used for a wideband (100 KC) relay system.

For a low capability SMS, a VHF coherent transponder is proposed for RARR measurements. This would be less accurate than an S-band transponder but would weigh less. The VHF transponder would consist of the telemetry transmitter and command receiver arranged to be coherent. The maximum sidetone frequency for ranging would be 20 KC.

When in its station keeping position, it is proposed to track the SMS by using the large parabolic antenna to track on the sensor data emissions. As an alternative or supplement, range and range rate equipment available anywhere within line of sight from the SMS can be used with the S-band beacon capabilities mentioned above. Another possible method is optical tracking with specially designed cameras such as the Baker-Nunn, ROTI or TPR units. However, these require clear weather at dawn or dusk for their operation.

### 5. Conclusions and Recommendations for Operating Communication Equipment

No development problems are anticipated in the obtaining of communication equipment for command, telemetry or tracking functions.

## D. ATTITUDE CONTROL AND STATION KEEPING

### 1. General

Preceding subsections have reviewed the study results for several SMS subsystems, mainly as to operations after arrival of the satellite on station. This subsection summarizes the investigations on alternate orbits, ascent to them, injection, perturbations and station keeping. It provides a brief review of the information on these subjects contained in Volume 4.

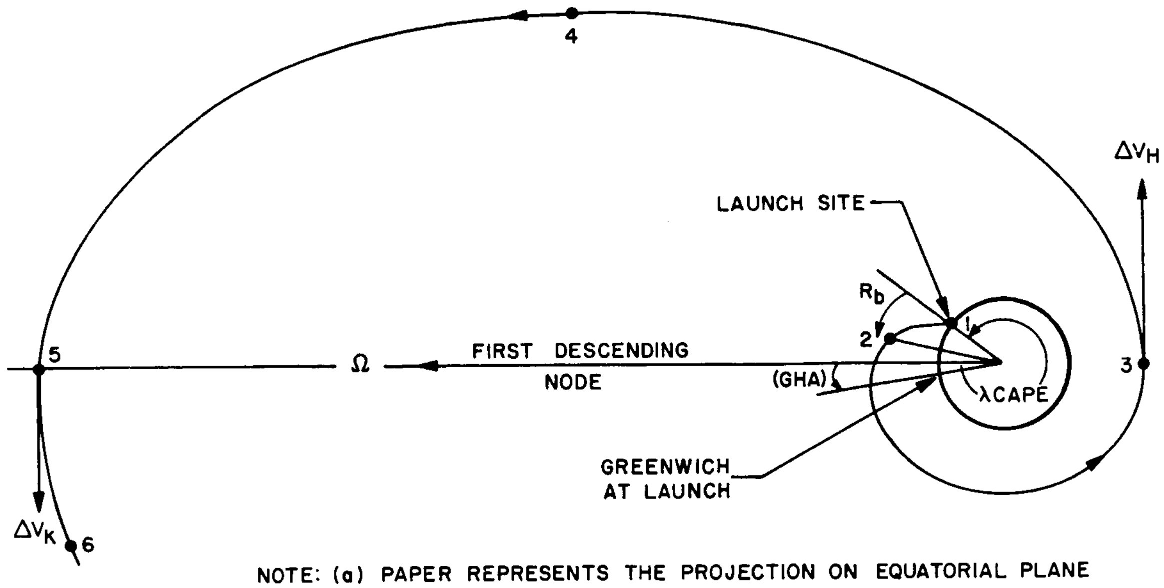
### 2. Ascent Phasing

#### a. Low Altitude Waiting

Ascent phasing techniques required to inject a payload into a 24-hr synchronous equatorial orbit located at 90°W longitude have been investigated. Booster vehicles with upper stages having restart capability (such as the Atlas-Agena and Atlas-Centaur) essentially can perform this mission within 6.4 hr from launch. This ascent path is characterized by a small off-East launch azimuth and a short wait time in an inclined parking orbit of 100 nm altitude. A Hohmann transfer, coplanar with the near circular waiting orbit, is initiated by a second burn of the Agena or Centaur stage at the first ascending node. Hohmann apogee corresponds to the 24-hr orbit altitude. At synchronous altitude, a solid fuel rocket imparts a 6030 ft/sec impulse which reorients the orbit plane and (with proper direction and timing) establishes the desired orbit and station. Figure 4-8 shows the ascent trajectory, and Figure 4-9 gives the associated ground track.

#### b. High Altitude Waiting

For boost vehicle systems lacking a restart capability in the upper stage (Thor-Ad, Thor-Delta), a maximum wait time of 2 weeks may be required to establish the 90° W longitude station. The powered ascent terminates at perigee of a Hohmann transfer trajectory. Unlike the low waiting trajectory,




NOTE: (a) PAPER REPRESENTS THE PROJECTION ON EQUATORIAL PLANE
(b) ALL ANGLES SHOWN IN A POSITIVE SENSE

1. LAUNCH
2. SECOND-STAGE PARTIAL BURNOUT AT 100-NM WAITING ORBIT (INCLINED 28.5°)
3. SECOND-STAGE IMPULSE APPLICATION FOR DEPARTURE FROM 100-NM PARKING ORBIT INTO HOHMANN TRANSFER ORBIT
4. VEHICLE IN HOHMANN ELLIPTICAL TRANSFER ORBIT (INCLINED 28.5°)
5. APOGEE KICK FOR REORIENTATION AND CIRCULARIZATION
6. VEHICLE IN 24-HOUR ORBIT

Figure 4-8. SMS Ascent Trajectory

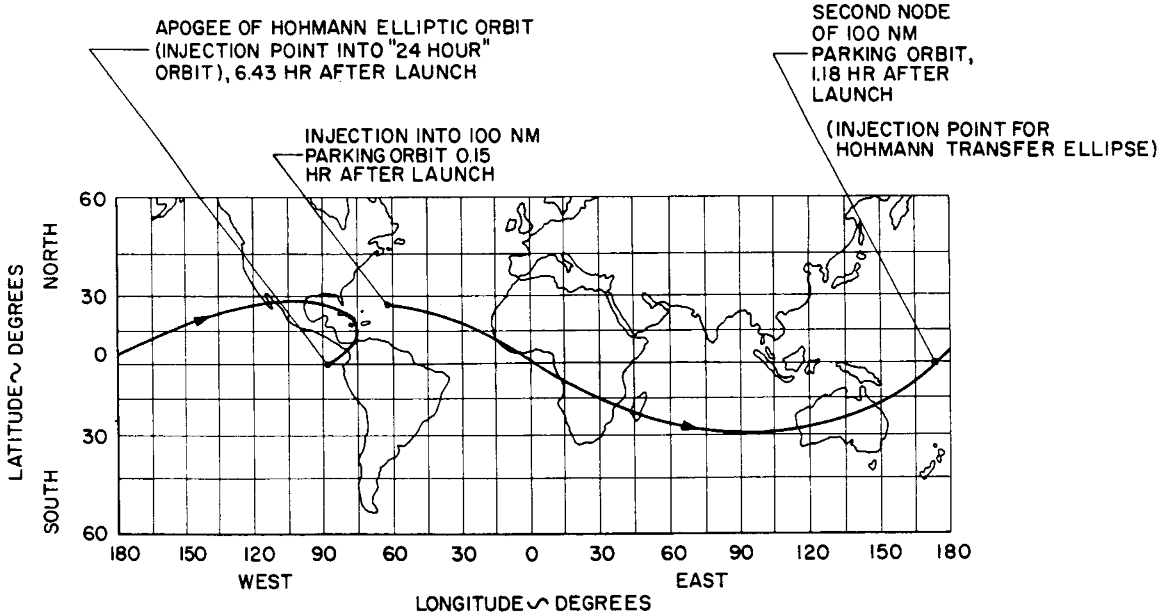



Figure 4-9. Earth Track of Low Altitude Ascent Wait Mode

the Hohmann perigee is not located at a node. An apogee kick velocity corresponding to 4653 ft/sec is applied in the orbit plane at 5.4 hr after launch. This establishes a high altitude waiting orbit inclined approximately 28.3° to the equator. This causes the satellite to drift eastward. The initial central longitude can vary between 20.5 and 38.5° E depending on the ascent trajectory. Two additional impulses totaling 199 ±39 ft/sec are deemed necessary for final circularization and for correction of the specified injection errors. In designing for the specified ±1000 sec random period error, propellant equivalent to 78 ft/sec may remain unused. Injection velocity errors greater than 39 ft/sec (corresponding to the period error of 1000 sec) would alter the above requirements; but the total required increase in characteristic velocity for the injection and circularization maneuvers would still be equal to the random injection velocity error.

### 3. Injection Error Correction

For the nominal injection errors given in the NASA contract (representative of the Centaur guidance system), the only ones requiring correction are period (1000 sec, corresponding to a mean motion error of -4.2° per day) and eccentricity (0.013). Both can be compensated for, by the application of circumferential (transverse) thrust, initiated between 5 and 24 hr after the apogee kick maneuver, and after accurate orbit determination and arrival at a favorable position in orbit. The off-station longitude error that accrues prior to the correction is considered as an initial condition for the station keeping mode.

Atlas-Agena guidance errors are larger than those cited above for Centaur. On the basis of SMS mission analyses, daily latitude excursions of ±5° are considered tolerable, with 2° preferred. Accordingly, corrections are needed only for the 3300 sec expected period error, corresponding to a semi-major axis error of 0.17 Earth radii, and for the 0.025 eccentricity. In general, these corrections can be accomplished efficiently by a single transverse thrust, using a hot gas propulsion system. Two transverse impulse maneuvers will be required when the semi-major axis error exceeds 0.06 Earth radii and the eccentricity error is close to zero, and when the former is near zero but the latter exceeds 0.009.

### 4. Perturbations

A satellite initially located at 90°W longitude in a synchronous, equatorial orbit will drift 75°W in 1.1 years, in the absence of station keeping. This longitude excursion results from the sectorial harmonics of the Earth's gravitational field. A longitude station keeping requirement is thus established. Lunar-solar perturbations can cause a maximum latitude excursion of ±0.75° after one year. This is well within the limits indicated by SMS mission requirements. No latitude corrections are required.

The foregoing results are not substantially altered for an orbit with modest inclination (under 30°). The longitude drift is reduced slightly, to 5° in 74 days when the inclination is 28.3° as compared to 70 days for no inclination.

### 5. Station Keeping

The characteristic velocity necessary to maintain a station position at 90° W longitude is approximately 7 ft/sec/year. This velocity requirement is independent of the number of discrete station keeping corrections made during a year's operation. Simple guidance logic is recommended, in which, ideally, transverse thrust is applied for a predetermined length of time when the satellite drifts off station by some preset angle. For example, two circumferential impulses totalling 7.4 ft/sec are sufficient to maintain a ±2° station longitude tolerance for 347 days after injection, as illustrated in Figure 4-10.

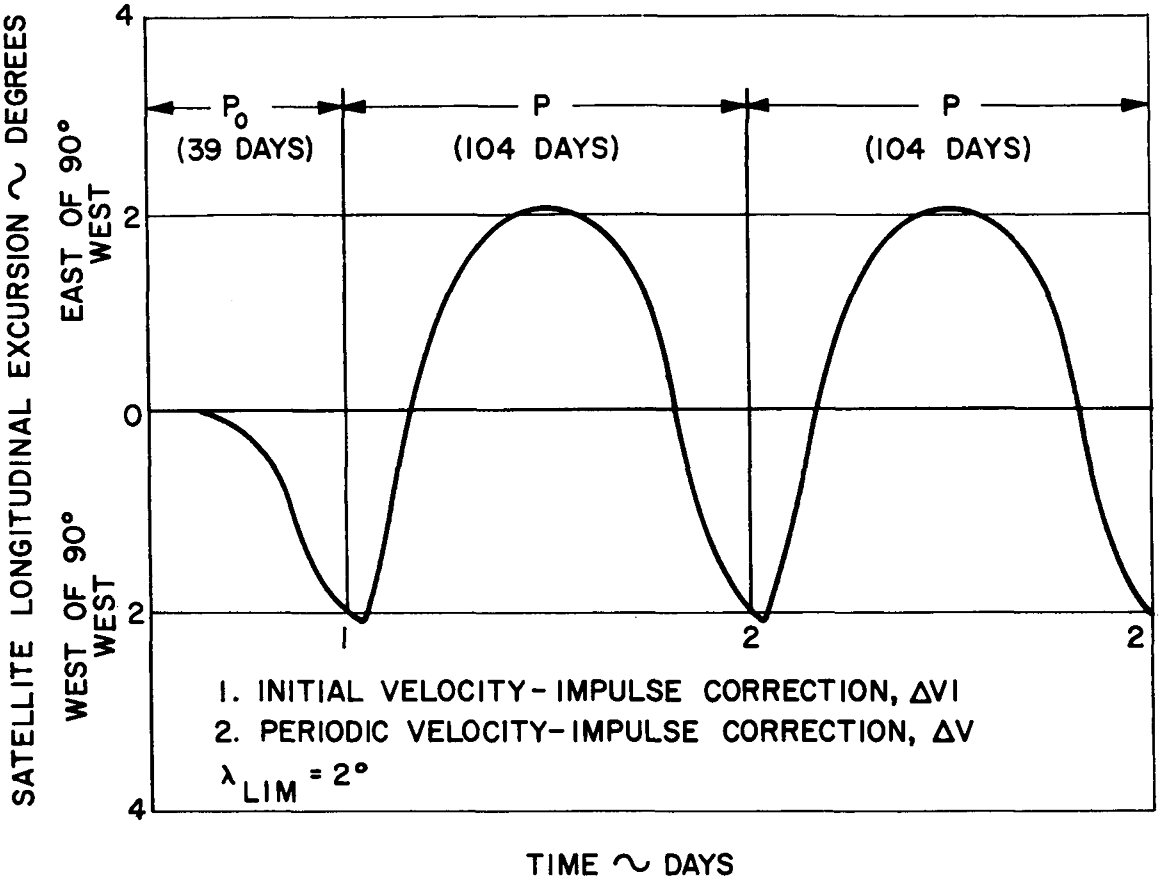



Figure 4-10. Approximate Station Keeping Pattern

Because of initial conditions that will exist on longitude and on its rate at the start of the station keeping mode, it will probably be advisable to control the thrusting interval by commands from the ground. The commands would be based on measurement of existing drifts past the control boundary.

6. Inclined Orbits

Certain other features of inclined orbits should be recognized for SMS applications. The first is the daily north-south figure-8 excursion about the nominal station on the equator. This can be used to extend the daily coverage area for meteorological observations. Second, there is a 14 percent reduction in the required total impulse, and hence in the mass of the apogee kick motor, such that, for an inclination of 28.3° an SMS of over 14 percent greater weight can be put into a synchronous orbit with any available booster vehicle.

7. Control of Attitude and Rates

The primary function of the SMS attitude controller after arrival on station is to provide the quality of attitude control and stabilization required to accomplish the basic meteorological mission in the presence of the expected disturbance torques. The block diagram in Figure 4-11 illustrates the generalized

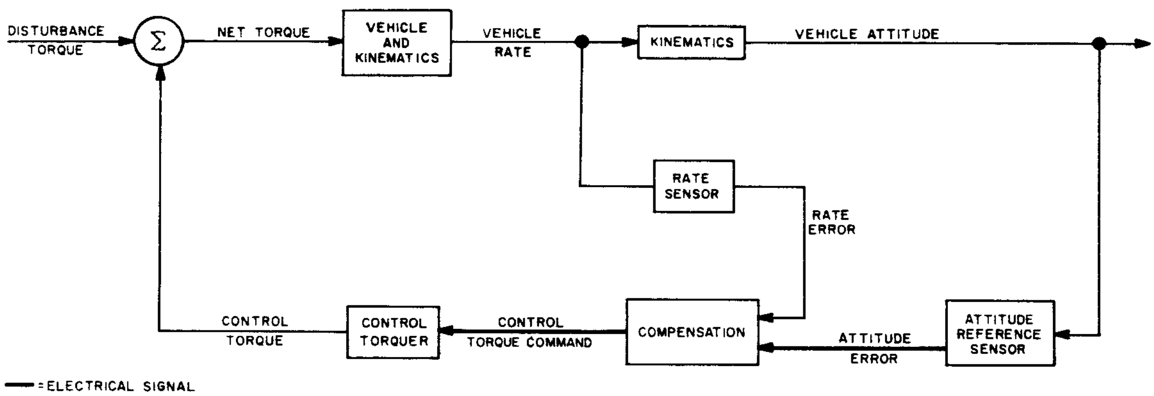



Figure 4-11. Descriptive Attitude Controller, Block Diagram

functions required for attitude control. Disturbance torques act on the vehicle, producing rate and attitude errors. These errors are transduced by attitude and/or rate reference sensors and, after appropriate signal compensation, used as commands to a control torquer that acts to counter the original disturbance torques.

8. Illustrative SMS System for Attitude Control

The attitude control system chosen for the 524-lb configuration is illustrated in Figure 4-12. This block diagram shows the integrated control functions of the 3-axis stabilized system. This particular 3-axis system is called the gyro compass control system and includes the following:

(1) Two variable speed reaction wheels

(2) One fixed speed wheel

(3) One 3-axis cold gas system

(4) Three rate gyros

(5) Four horizon sensors

(6) Control logic for various modes

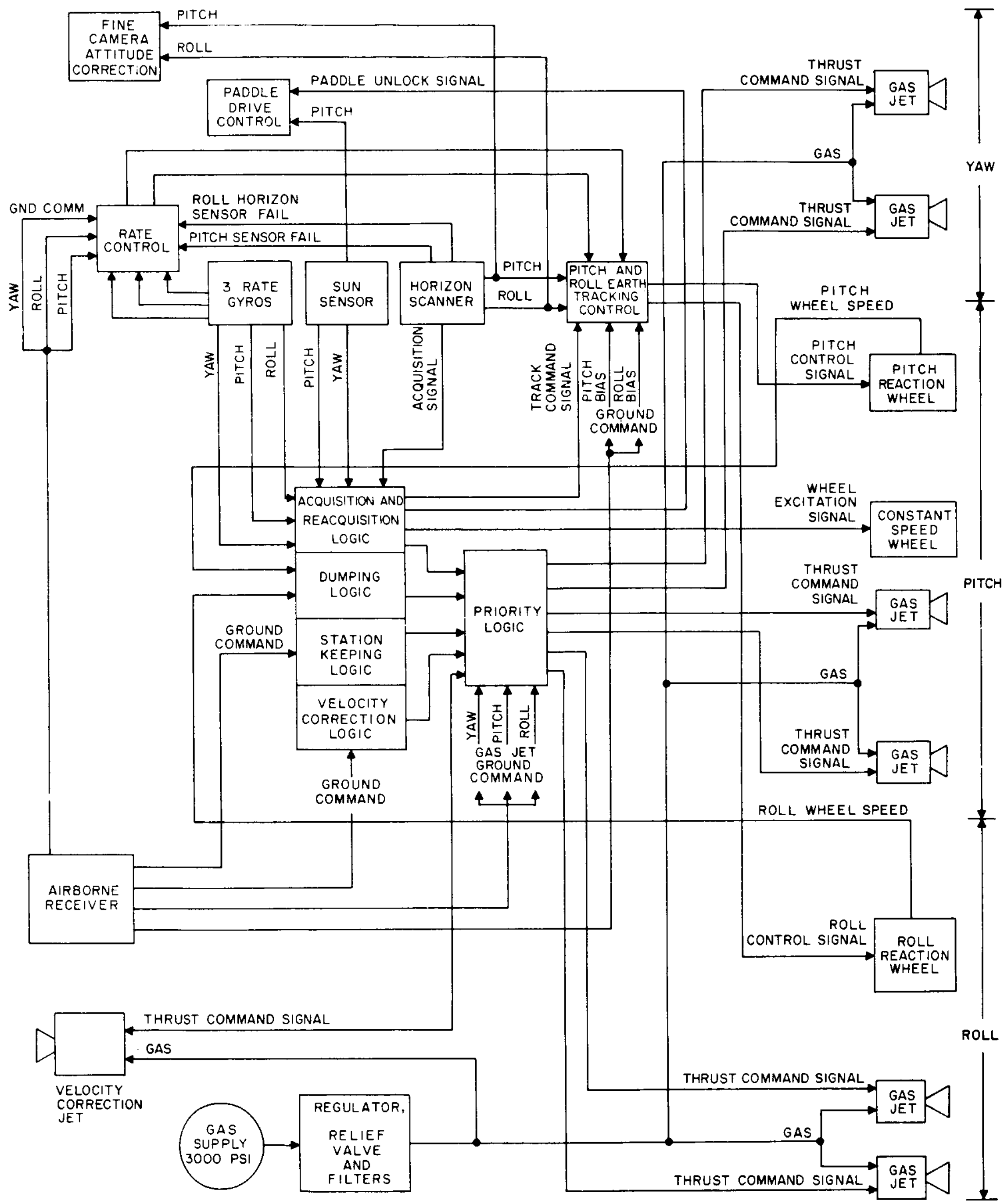



Figure 4-12. Control Systems for 3-Axis Stabilized Satellite, Block Diagram

The modes of operation provided by this system configuration are as follows:

(1) Automatic Earth acquisition and reacquisition

(2) Automatic Earth tracking

(3) Command control of initial velocity error correction

(4) Command control for station keeping

(5) Automatic or command control of gas system for reaction wheel momentum unloading

(6) Command control of pitch and roll bias command

(7) Command rate control mode

(8) Automatic solar paddle drive control

A brief description of each of the operational modes and control logic functions is provided in the following paragraphs.

a. Control Logic Functions

1) Priority Logic. The priority logic block indicated in Figure 4-12 performs the function of operating the cold gas system on a priority basis. That is, the Earth acquisition and reacquisition mode has priority over momentum dumping, velocity correction, and stationkeeping, and operates the cold gas system in a rate mode or a Sun orientation mode. Momentum dumping has priority over the velocity correction and stationkeeping functions which are locked out if a dump signal exists. Velocity correction has priority over stationkeeping, but since both of these are ground commanded, the ground complex will know that initial velocity corrections must be performed prior to stationkeeping. Ground command signals are available to the priority logic block which provide the capability for command dumping.

2) Acquisition and Reacquisition Logic. Acquisition and reacquisition logic provides for the control of the cold gas system for final active despin, Sun-orientation, and rotation about the Sun line which are fully described in Section 2. F of Volume 4. The acquisition logic provides yaw, pitch, and roll rate signals for active despin and for subsequent rotation (roll) about the Sun line to effect Earth acquisition by the horizon scanner. It also accepts Sun sensor pitch and yaw signals for the previous Sun-orientation phase and an acquisition signal from the horizon scanner. This horizon scanner signal activates the pitch and roll Earth tracking control system, provides excitation to the constant speed wheel for initiating yaw alignment, and provides a solar paddle unlock signal.

3) Dumping Logic. This block serves the purpose of providing signals to the cold gas valves for the removal of pitch and roll reaction wheel angular momentum. Inputs are reaction wheel tachometer signals which activate and deactivate the dumping logic on the basis of reaction wheel speed.

4) Injection Velocity Correction Logic. This logic selects the proper cold gas nozzles for injection velocity correction depending on the sense transmitted via ground command. The duration of velocity correction depends on the duration of the ground command signal which is based on a measured orbital velocity error or longitude drift rate.

5) Station Keeping Logic. This logic activates the cold gas nozzles for correction of velocity or mean motion errors due to orbital perturbations. Since the major perturbation is due to the Earth's equatorial ellipticity, this velocity error will always be one-sided; hence, corrective thrust will always be applied in the same direction, opposed to the direction of motion. The ground command signal is therefore on-off in nature and does not require sign intelligence.

b. Operational Modes

1) Automatic Solar Paddle Drive Control. This mode provides for the alignment of the solar paddles with the Sun. The Sun sensor provides a continuous error signal to the paddle drive servo. The drive servo thus maintains a paddle rate relative to the spacecraft body. This rate is equivalent to the angular velocity of the spacecraft about the Earth combined with the angular velocity of the Earth about the Sun.

For purposes of storage within the launch vehicle the paddles must be folded and are not deployed until after apogee kick. Following deployment they must be locked in a position where the paddle surface is parallel to the body yaw-pitch plane. Thus, during the Earth acquisition mode (following Sun-orientation of the roll axis) when the vehicle is rolled about the Sun line, the solar paddles will be fully illuminated.

2) Command Rate Control. An external ground command rate mode is provided in the event horizon sensors fail, and also to provide stabilization at extremely low rates for high resolution pictures. Attitude feedback for this mode, in the event of horizon sensor failure, is by means of photographs of the Earth disc, and noting its displacement from frame center.

For the case where horizon sensors were operative, the system would revert back to the normal tracking mode after completion of high resolution pictures.

3) Command Control Pitch and Roll Bias. A ground command exists which can supply bias signals to compensate for horizon sensor bias errors. To ascertain that an error actually exists it would be necessary to analyze Earth disc pictures and establish the magnitude of the pitch and roll errors.

Other modes of operation are described in the control logic section of Volume 4. Acquisition and Earth tracking are fully discussed in Sections 2. F and 2. E of Volume 4.

4) Attitude Reference Sensor Characteristics. The performance characteristics of the attitude reference sensor, considered to be an infrared horizon scanner, are of primary importance, when assessing control loop

performance. Horizon scanner noise and deadzone characteristics were considered. For a scanner of the OGO type discussed previously, a white noise of 0.02 degree RMS value is expected. It is found that this noise gives rise to RMS pitch and roll rates of 0.003 degree/sec. Due to the system dynamics no significant yaw rate is induced. The RMS yaw angle was also negligible, while the RMS roll angle was about 0.02 degree. The RMS pitch angle could also be expected to be about 0.02 degree.

In the gyro compass system, the scanner deadzone gives rise to a complex, low amplitude oscillation. However, the effect from the oscillation would not be unacceptable. Analytical and analog computer studies support predictions of pointing accuracies of 0.5 degree for all axes and rate stabilization levels of 0.003 RMS degree/second.

9. Alternative 3-Axis Systems

A discrete-continuous system, typified by a 3-axis reaction wheel-cold gas system, which does not utilize the gyro-compassing feature, was not chosen for the SMS because of stability problems associated with gyroscopic roll-yaw coupling. In the SMS orbit the solar pressure torque will require more frequent dumping of reaction wheels or larger wheels, both of which add weight to the system.

In Section 2.E of Volume 4, these and other 3-axis systems are analyzed. They differ mainly in the type of torque device used, or in the implementation of control logic. It is apparent that a major problem exists in the area of attitude control sensors for the Earth oriented satellite mission when a high degree of pointing accuracy is required in all axes.

An area for profitable future study would be the investigation and exploitation of high accuracy sensors such as Sun sensors and star trackers for the Earth oriented satellite mission. It is clear that existing Earth horizon sensors are undesirable for an Earth oriented mission requiring highly accurate pointing and/or rate stabilization. If high precision sensors as used for inertially fixed satellite missions could be used in a simple reliable manner for the SMS mission, an obvious weak link will have been eliminated.

Many of the torquing devices investigated in this study would be applicable to the high performance mission. Indeed, the basic problem area is that of attitude sensing.

A likely control system for the high accuracy mission would employ a Polaris star tracker for roll and yaw reference, and a single-axis, gimbaled Sun tracker which tracks the Sun in a plane containing the vehicle pitch axis and the Sun. Horizon scanners would not be used during Earth tracking, but would be of use during the acquisition phase. Nonrotating torque devices would prove more suitable than those which produce inter-axis coupling.

## 10. Conclusions and Recommendations for Attitude Control and Station Keeping Equipment

(1) The preferred stabilization system for SMS is a 3-axis stabilized system using reaction wheels for roll and pitch stabilization and a constant speed wheel in a gyrocompassing mode for yaw control. The reaction wheels would be unloaded by either a high pressure or subliming cold gas system.

(2) The cold gas system used for unloading the reaction wheels can also be used to perform the station keeping function.

(3) The use of a subliming cold gas system, rather than high pressure cold gas, will provide a simpler gas system and a saving in weight.

(4) The above system for attitude control and station keeping can be designed and built now.

(5) The limitation on the tolerance in attitude control is the 0.5 degree accuracy of the infrared horizon sensors. If greater accuracy becomes necessary due to change in sensor requirements, a star tracker using the North Star (Polaris) and a Sun tracker could be used in lieu of horizon scanners.

## E. POWER SUPPLY SUBSYSTEM

### 1. General

The principal subsystems for which electrical power is to be provided have been reviewed. This subsection surveys the alternate types of power systems which were considered. Section 3 of Volume 5 provides a more detailed parametric study of the power supply problems.

As examples of power requirements, the electrical loads established by the subsystems involved in the three illustrative SMS configurations cited earlier are listed in Table 4-1.

### 2. Preferred Power Systems

Table 4-2 presents the weights of the power systems chosen for the three illustrative configurations. The lightest one was stabilized by spin, so that solar cells were mounted on cylindrical external surfaces. The other two utilized 3-axis stabilization; the solar cells were mounted on paddles oriented toward the Sun by rotation about an axis parallel to that for spacecraft pitch, hence parallel to that of the Earth's rotation.

Other features of these illustrative systems and two other configurations are analyzed in Volume 2.

### 3. Parametric Power Source Study

The pertinent characteristics of the various power generation and energy storage devices that were studied for application to the SMS are summarized in Table 3-11 in Volume 5. The approximate power level and the mission duration most suitable for each type of system are shown. Also given there are the relative weights of each of the systems in their present state of development.

For satellites on brief missions, various forms of batteries can serve as the electrical power supply. Long-life satellites require considerably increased levels of energy, and the weight of electrochemical devices becomes excessive. Solar energy and photovoltaic converters (silicon solar cells) can be used in this case, for converting solar energy into electricity, with the storage of energy being accomplished by a secondary battery. On the basis of the current state of development and on demonstrated cycle life, nickel-cadmium batteries appear to be the best overall choice. There is evidence that silver-cadmium cells cannot, in general, accept charging rates as high as nickel-cadmium cells. Silver-zinc cells do not warrant strong consideration, at this time, for the synchronous orbit energy storage application because of a relatively short cycle life.

Fuel cells, when they are more fully developed, could become competitive with battery and solar cell systems. At present, however, fuel cells such as the hydrogen-oxygen type would be applicable mainly for missions in which several hundred watts are required for several days. For higher power levels, at which solar cells become too cumbersome, nuclear power systems offer the best prospects as a power generating source. Three SNAP nuclear reactor projects are now programmed for early orbital flight testing. These are the 500 watt

TABLE 6-1
ELECTRICAL LOADS FOR THREE SMS CONFIGURATIONS

| Systems | 218-1b Configuration | | | 524-1b Configuration | | | 775-1b Configuration | | |
|---|---|---|---|---|---|---|---|---|---|
| | Earth Acquisition (Watt-Hours) | Orbit Operation (Watts) | | Earth Acquisition (Watt-Hours) | Orbit Operation (Watts) | | Earth Acquisition (Watt-Hours) | Orbit Operation (Watts) | |
| | | Avg. | Peak | | Avg. | Peak | | Avg. | Peak |
| Meteorological Sensors | 0 | 7 | 15 | 60 | 77.1 | 81 | 60 | 85 | 144 |
| Communications | 132 | 29.5 | 42 | 360 | 50 | 75 | 360 | 55 | 83 |
| Data Handling | 134.4 | 24.4 | 24.4 | 186.6 | 33.1 | 33.1 | 186.6 | 36.4 | 36.4 |
| Attitude Control | 21.1 | 31.7 | 56 | 79 | 105 | 241 | 79 | 105 | 241 |
| Power Supply | 42 | 7 | 7 | 42 | 7 | 7 | 42 | 7 | 7 |
| Totals | 329.6 | 99.6 | 144.4 | 727.6 | 272.2 | 437.1 | 727.6 | 288.4 | 511.4 |

TABLE 6-2
POWER SYSTEM WEIGHTS FOR THREE SMS CONFIGURATIONS

| System | System Weight in Lb 218-lb Configuration | 524-lb Configuration | 775-lb Configuration |
|---|---|---|---|
| Total Power Subsystem | 80.7 | 127.7 | 172 |
| Primary Battery | 8.5 | 18.2 | 0 |
| Secondary Battery | 6.2 | 21.5 | 67 |
| Solar Cell Array | 54 | 73 | 90 |
| Regulator - Charge Selector | 12 | 12 | 12 |
| Solar Paddle Drive | 0 | 3 | 3 |

SNAP-10A (1964), the 3 KW SNAP-2 (1965) and the 30/60 KW SNAP-8 (1967). The radioisotopic SNAP program has been oriented toward supplying the need for small, compact, lightweight power supplies for space applications. Small nuclear power supplies in general are either too heavy or have too short a life for SMS application. For example SNAP-1A has a design life of 1 year but weighs 200 lb for 125 watts output. A space probe generator that produces 100 watts for a 77 lb weight has a life of 6 months.

For power levels below 10 KW, nuclear reactor systems are several times heavier than solar cell-battery supplies because of shielding, cooling, and other requirements. They cannot be reduced in weight beyond a certain lower limit, without decreased power output requirements. At the 500 watt level a reactor system weighs approximately 950 lb while the solar cell-battery system weighs approximately 250 lb. Nuclear systems do have the advantage of being relatively impervious to the hazards of space environment such as Van Allen radiation, solar flares, and the impingement of micrometeors. In addition, nuclear sources eliminate the need for secondary batteries and Sun orientation requirements.

Solar thermionic systems appear to be a good choice when power levels from 0.1 to 20 KW are needed for no more than several weeks. This is particularly true at power levels above 1 KW, when the solar cell panel arrays of the solar voltaic systems become excessively heavy as compared to the concentrators used in solar thermionic systems. The disadvantages of thermionic systems are that they have limited life at present and their solar concentrators must be focused with a considerable amount of accuracy. Chemical power systems, too, appear to be most useful as auxiliary power sources when high power is required for short duration missions lasting several hundred hours or less, during which power levels of 2 KW or greater are required.

4. System Design

The power level of the satellite affects two design factors; the stabilization method, and the solar-cell array design. Spacecraft requiring less than one hundred watts are likely to be spin stabilized. This reduces the complexity of the stabilization subsystem, but requires an increase in the number of solar cells by a factor of $\pi$.

At medium power levels a choice exists as to whether or not the solar-cell array is an oriented or nonoriented system. Flat panels are generally used in oriented solar power systems. The exact layout is determined by payload shape, storage space, and erection mechanisms. In a nonoriented system such as the spin stabilized spacecraft, the solar-cell area is distributed about the satellite surface proper. In a nonoriented system, therefore, a large portion of the active surface will generally not be fully utilized. Figure 4-13 compares solar-cell distributions for the various SMS configurations. It shows the collector area penalty, i.e., the ratio of the solar-cell area required for a nonoriented system to the surface area required for an oriented system. Volume 2 discusses in detail the criteria involved in selecting a specific configuration.

The 524-lb satellite is a 3-axis stabilized, Sun-oriented system operating in a synchronous orbit about the Earth for the life duration of one year. Solar-cell panels are used; they are designed to provide the average power re-

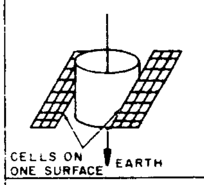
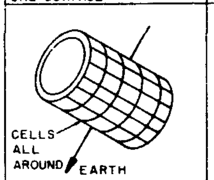
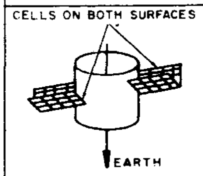
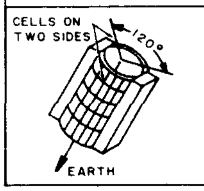
| CONFIGURATION | TYPE OF ARRAY | AREA PENALTY FACTOR | SOLAR CELL AREA FACTOR |
|---|---|---|---|
|  CELLS ON ONE SURFACE; EARTH | ONE-AXIS ORIENTED ARRAY | 1 | 1 |
|  CELLS ALL AROUND; EARTH | SPIN STABILIZED | $\pi$ | $\pi$ |
|  CELLS ON BOTH SURFACES; EARTH | FIXED ARRAY (90° PANELS) | 1.66 | 3.32 |
|  CELLS ON TWO SIDES; 120°; EARTH | BODY MOUNTED | 3.28 | 3.28 |

Figure 4-13. Solar Cell Distribution for Gravity Gradient Satellite

quired for the vehicle mission. Batteries are used to meet acquisition, shadow, and peak load requirements.

The solar cell characteristics vary approximately with the cosine of the angle of incidence of solar radiation. The effect of temperature on current is relatively small, but that of power (and voltage) decreases rapidly with increasing temperature. It is usual to allow 15 percent additional solar-cell array capacity to account for uncertainties in orbit precession and attitude variations.

At the present time, silicon solar cells of 12 percent conversion efficiency are available in quantity. The design takes into consideration the degradation of solar-cell efficiency with, say, ±100°C temperature variation from 0°C. The silicon-cell efficiency goes down about 50 percent with 100°C rise in temperature. Thus, it will be necessary to assume that the efficiency of solar energy conversion is only 6 percent though the cells used are of the 12 percent efficiency type (at 0°C). Since the solar power available at the Earth, and with slight variation in the synchronous orbit, is 130 watts/square foot, the solar-cell output will be 6 percent of 130 or 7.8 watts/square foot. Reducing this figure 4 percent to include the loss of solar-cell area due to contact shingling and grid lines, it can be assumed that the power density of the solar-cell array is 7.5 watts/square foot.

Electron flux at the synchronous orbit altitude due to the artificial

Van Allen belt is negligible. Protection from Van Allen radiation and micrometeoroidal damage at this altitude may be provided by a 6-mil quartz cover. It is advisable, however, to increase the solar-cell area by an additional 30 percent to allow for solar-cell degradation due to solar flares.

The required output voltage establishes the number of cells which must be connected in series and the load current requirement establishes the number of parallel banks of series strings which must be provided. Though the output voltage and current requirements control the overall number of series and parallel cells, the solar cells may be interconnected into series-parallel groups, for increased reliability. The exact configuration must be deduced from a thorough statistical analysis of the probability of multiple failures and their resultant effect on array characteristics.

### 5. Battery Considerations

Since battery life is affected by the depth of the charge-discharge cycle and by overcharging, accurate control of solar cell voltage is essential. For the SMS orbit, the shadow period is about 70 minutes out of 24 hours during the maximum shadow period. With a slow charge rate, which can be used with negligible cell damage, a 65 percent depth of discharge is possible for the battery to account for loads while the satellite is in the Earth's shadow. Since the power profile is such that there is a rapid occurrence of peak drains on the battery during normal operation, i.e., over consecutive half-hour cycles for extended periods of time, the secondary battery must be of such capacity as to provide that the depth of discharge, at the peak drain during normal operation, will not exceed 10 percent of the secondary battery capacity. At the same time, the secondary battery capacity should be of such magnitude that the depth of discharge due to shadow load will not exceed 65 percent of the secondary battery capacity. It will therefore be necessary to use a variable type charge rate circuit rather than a slow constant current taper. This type of charging will, in effect, charge the battery at a rapid rate immediately after a shadow period and at a slower rate during normal operation.

The cell array-battery combination will feed the various subsystems through a voltage regulator circuit. Those subsystems requiring uncommon power, i.e., increased or decreased voltage or AC power, etc., will have their individual conversion systems at or within the respective equipment.

### 6. Conclusions and Recommendations for Power Supply Equipment

(a) The recommended power supply uses silicon solar cells and nickel cadmium batteries, both of which are now available.

(b) The spacecraft design provides a margin of about 18 percent excess solar cells, computed on a 6 percent cell efficiency, above the average power requirement.

(c) The development of batteries with long life will increase the reliability of the spacecraft.

(d) The development of light weight nuclear power supplies of long life would be advantageous for SMS application.

## F. SPACECRAFT CONFIGURATION AND STRUCTURE

### 1. General

The combination, arrangement, and integration of many subsystems into a practical spacecraft, within weight limits imposed by available booster vehicles, present many problems. Some solutions are illustrated by Figures 3-2, 3-3, and 3-4. Figure 3-1 and Tables 3-1 and 3-2 indicate the weight distributions and performance estimated for these three illustrative configurations.

Some of the considerations leading to these configurations are summarized in this section and are presented in Volume 2 in more detail.

### 2. Configurations

The final configuration of any satellite is the result of many compromises, usually made in favor of the most important subsystems. In the case of the SMS, the most important subsystems are the sensor equipment and the closely related data transmission subsystem.

Aside from the limitations of the sensors themselves, the basic controllable parameter is platform stability. Platform stabilization has been surveyed from different angles, including a positive directed control, the use of naturally available forces, and the use of a vehicle motion that in itself supplies stability. The use of each of these principles leads to a distinct final configuration.

Positive directly controlled stabilization leads to the 3-axis stabilized satellite. This satellite, depending on the attitude-sensing system and on controlled reactions, can be held to stability levels determined by the sensors and controllers. Use of horizon scanners in the control loop is a limiting factor on pointing accuracy, which is approximately $\pm 0.5°$ for the control systems selected. Platform oscillation rate is much more important, and can be held to $10^{-4}$°/sec if the horizon scanner is optimized or removed from the control loop during picture exposures.

Use of the natural force of gravity gradient results in a satellite in which the moment of inertia is of prime importance. This satellite tends to point toward the Earth and can be sized to point quite accurately, thereby eliminating need for a horizon scanner. Its oscillation rates must be controlled and damped, requiring an active damping system of complexity comparable to that of the 3-axis stabilized satellite or a highly effective passive 3-axis damping device. The problems of orienting and erecting this satellite in space to attain the moments of inertia required tend to overshadow its superior natural stability.

Spin stabilization is a method of using the revolving motion of a body to maintain stability. The gyroscopic characteristics of the spinning satellite tend to hold the spin axis fixed in inertial space. The fact that the satellite is necessarily symmetrical due to the balancing requirement tends to minimize unbalanced forces caused by solar flux. Thermal control is vastly simplified, since the satellite is basically isothermal. The spin motion provides a semi-active thermal control. Resistance to turning of the spin axis can be controlled by rotational speed. However, the stability of the spin axis in relation to the Earth does not produce platform stability for the sensors, since they are revolving at the spin rate and view the Earth intermittently as it flashes by. Image motion compensation for the optics in this configuration presents problems. Studies of the three types of configurations have led to the conclusion that the 3-axis stabilized satellite offers the best solution for the SMS mission.

### 3. Arrangement Criteria

The arrangement chosen for the 524 lb configuration is a logical outgrowth of the various subsystem criteria and mechanical problems. The circular section is a development of the launch vehicle spin table and of the need to house the circular apogee motor. The motor is buried in the satellite to reduce the adapter size and to minimize any eccentric loads introduced by the motor when fired. The hollow conical area also provides required space, access and motion for use of an air bearing to test and check the attitude control system. Solar paddles are located at the center of gravity of the satellite to minimize disturbances caused by solar pressure. The disposable load, attitude control gas is also located in the plane of the cg, so that use of the gas will not affect satellite balance. The circular section terminates in a structural bulkhead a short distance below the cg.

The circular section houses the equipment in the satellite that is not likely to change with varying missions. Telemetry, attitude control, power supply are among these basic items.

Mounted on the bulkhead closing the circular section are the meteorological sensors and the communications equipment. Enclosing them in this particular configuration is a rectangular box that serves as structure and thermal housing. Since a wide variety of sensors is proposed, and may be used at a later date, this arrangement is thought to provide the necessary flexibility for installation.

Passive thermal control will be used on the satellite, using oriented panels, specific finishes, insulation, heat sink material and local heaters. There are no moving parts for thermal control.

Solar paddles are oriented about the pitch axis. It has been necessary to fold them on two hinges in order to accommodate the necessary area within the Agena nose cone limits.

### 4. Structural Considerations

The proposed structure is a logical arrangement and can by application of careful design be kept within the weight limit. Vibration and high accelerations imposed by the launch vehicle during ascent, and later the loads imposed by the apogee kick motor, combined with the centrifugal loads due to spinning, must be carefully evaluated. Obviously, a complete test program will have to be accomplished to prove out the final configuration.

### 5. Conclusions and Recommendations for Structures

The structure must be considered a problem area. Every effort must be made to minimize weight; therefore, structure may not be over-designed. Vibration and critical frequencies often induce loads not foreseen in design stages. The following items must be designed and thoroughly tested:

(1) Passive despin system must be tested in 1 g condition in such a way as to accurately simulate the deployment in space under conditions of weightlessness.

(2) A reliable solar paddle and electrical slip ring assembly must be designed and tested.

(3) Solar paddle folding and release mechanism must be designed and proved.

(4) The satellite must be statically and dynamically balanced at about 100 RPM for the coasting phase during ascent to synchronous altitude.

It is to be noted that these are all rather straightforward engineering problems. Their solutions do not require uncertain developments or technological breakthroughs.

## G. THERMAL CONTROL

### 1. General

Thermal control problems are best attacked after the subsystems, subsystem arrangement, and satellite configuration are defined.

For all of the illustrative SMS configurations studied, with weights-in-orbit from about 200 to 800 lb, passive thermal control systems were found to be the best choice. These are systems involving no moving parts and no moving fluids. With them, the temperature control goal of 25±10°C can be achieved. Weight penalties are minimized by structural modifications which provide conductive paths and space radiators.

### 2. Scope

Considerations, analyses, and solutions of the overall problem of thermal control of the SMS are presented in Volume 5. The totality of energy sources and environmental conditions affecting the spacecraft in orbit are analyzed and evaluated.

Analytic and numerical solutions to the thermal interaction equations among the Sun, the Earth and the SMS are given for various SMS shapes. The umbra and penumbra shadow characteristics are shown as a function of calendar time.

Generalized thermal-balance differential equations are presented for three basic configurations; the spin-stabilized, the 3-axis stabilized, and the gravity gradient stabilized configuration. Satellite temperatures are shown as a function of time, surface characteristics, orientation, and internal power density.

Analyses are presented for thermal control by purely passive means, self-oriented space radiators (shadow boxes), energy storage devices, surface heating elements, shutter systems, thermal switches, forced convection loops (cold plate), vapor, gas and absorption cycle refrigeration systems, and thermoelectric cooling as applied to the SMS. Comparisons of weight, complexity, and integration problems of the various thermal control systems are shown. Analyses and solutions to the problems of infrared sensor cooling and solar cell thermal protection are presented.

3. External Heat Loads

The temperature-control problem for the SMS differs in a number of ways from that of a conventional, low-altitude satellite. The thermal control of the SMS is importantly influenced by the fact that the SMS is in the Earth shadow less than 1 percent of the time (74.7 hours in a year). The corresponding shadow times for conventional, low-altitude satellites can be as high as 50 percent, depending on the hour and day of launch. In addition, the shadow periods of the SMS do not occur at symmetrical intervals. For two 5-month periods each year, the SMS is constantly in sunlight. During March and September of each year, the satellite enters a daily shadow pattern with shadow periods ranging from zero to 65.95 minutes/day in the umbra and from zero to 24.1 minutes/day in the penumbra as shown in Figure 4-14. The umbra is the conical total shadow projected from the Earth on the side opposite the Sun. The penumbra is the partial shadow between the umbra and the full-light region. In the penumbra, the light of the Sun is only partially cut off by the Earth. The time the SMS is in the penumbra is only 13.35 percent of that of the umbra (9.9 hours/year).

Another important consideration in the temperature control of the SMS is the number of external heat sources that contribute significantly to the thermal environment of the satellite. In contrast to low-orbit satellite external thermal sources, the only external thermal source of any significance in the case of the SMS is the Sun. As is shown in Volume 5, the terrestrial and reflected (albedo) radiations at synchronous altitude (22,240 statute miles) are 2.16 percent and 1.55 percent of their respective near-Earth values and only 0.323 percent and 0.62 percent of the solar flux, respectively.

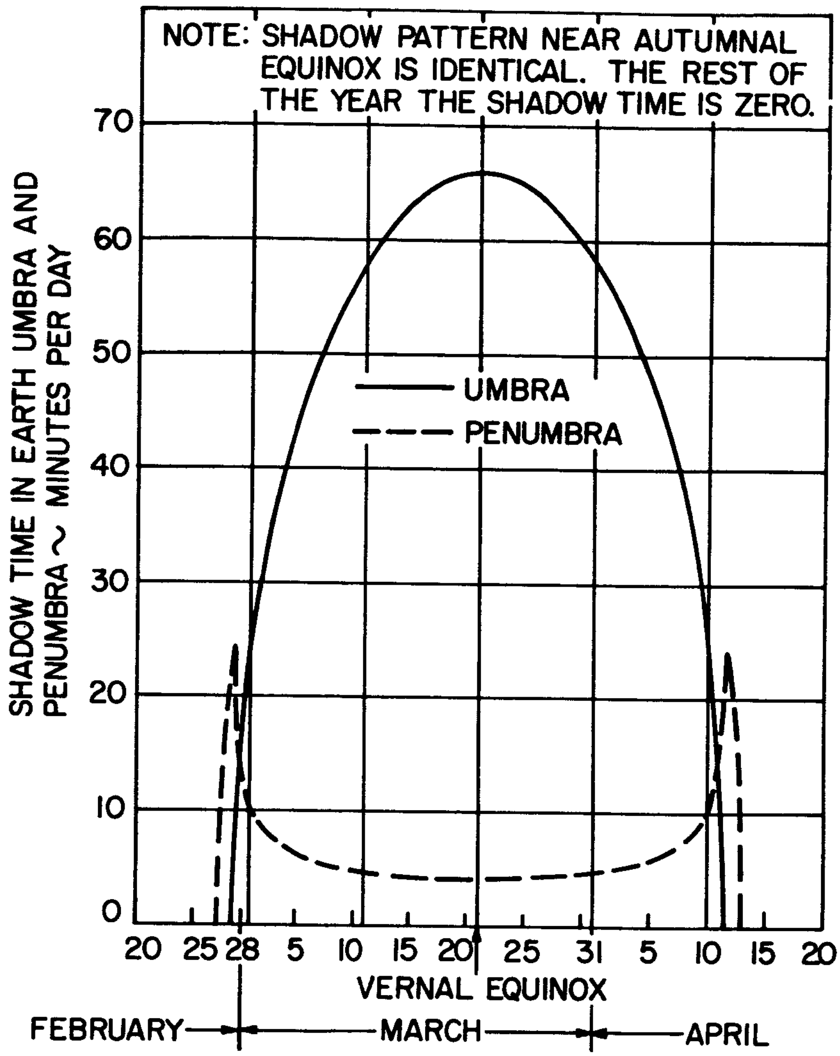



Figure 4-14. Umbra and Penumbra Patterns for a Synchronous Equatorial Satellite

The SMS thermal environment due to solar radiation is different for the spin-stabilized configuration and for the 3-axis stabilized configuration. The histories of the solar heat input to the 3-axis stabilized and the spin-stabilized satellite configurations are shown in Figures 4-15 and 4-16, respectively.

The long orbital period of the SMS is another significant factor in the thermal analysis. Low-altitude satellites have orbital periods in the order of 1.5 to 2 hours. With orbital periods of this order, thermal inertia can be fully utilized to dampen inherent orbital temperature fluctuations. With an orbital period that is 12 to 16 times as large (24 hours for the SMS) the additional degree of freedom of building in thermal inertia into the satellite is practically nonexistent.

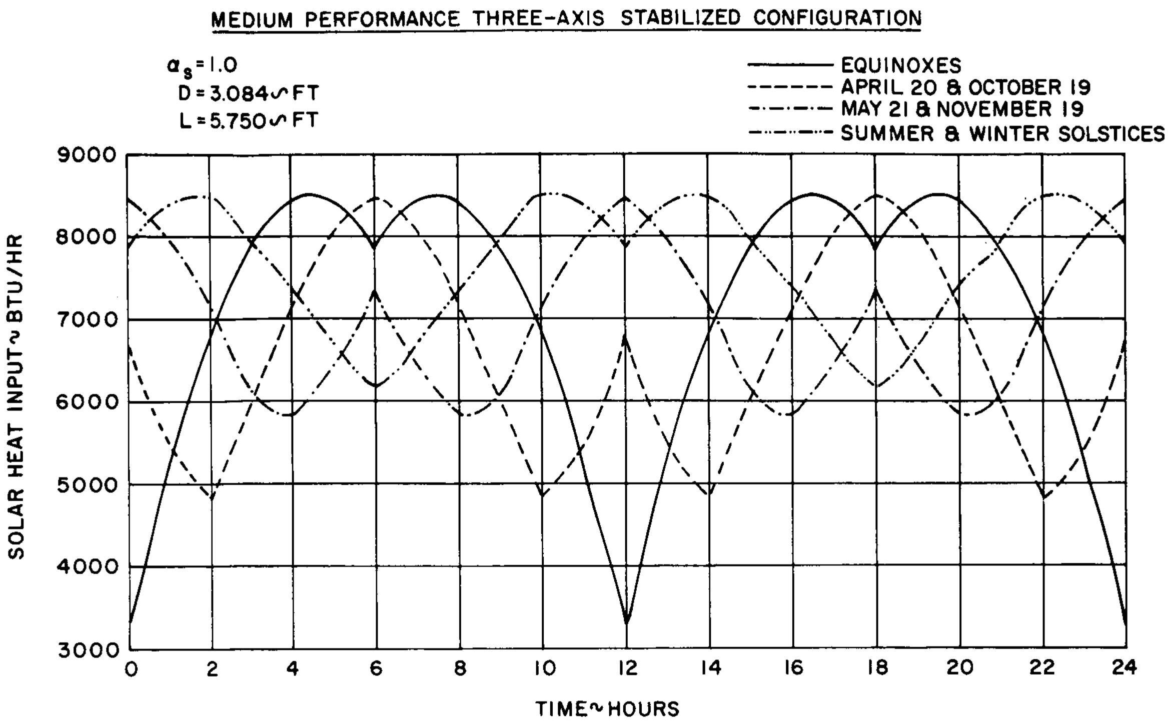



Figure 4-15. Variation of Solar Heat Input with Time of Day and Seasons

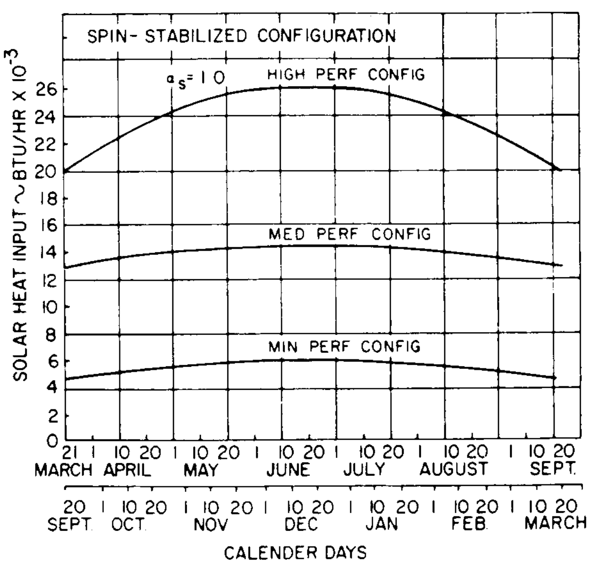



Figure 4-16. Variation of Solar Heat Input with Calendar Time, Spin-Stabilized Configuration

4. Thermal Analyses

The temperature of the SMS is a function of the following factors:

(1) The external and internal energy incident on the surface.

(2) The absorptivity, emissivity, transmissivity, reflectivity, thermal conductivity, and specific heat of the surface.

(3) The orientation of the satellite relative to the incident energy.

(4) The surface geometry.

(5) The energy emitted by the satellite surface.

(6) The thermal capacity of the structure and equipment.

Thermal analyses that take all of these factors into account are presented in Section 4 of Volume 5. The temperature histories for the spin-stabilized minimum, medium, and maximum performance configurations are shown in Figure 4-17.

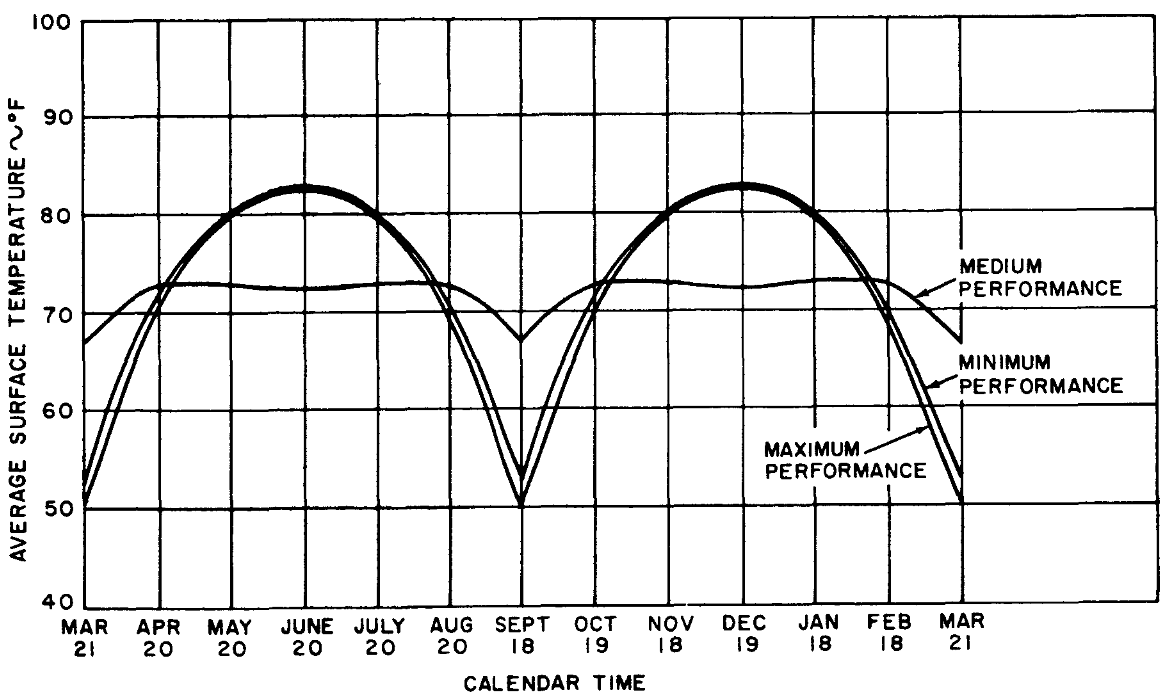



Figure 4-17. Average Surface Temperature for Spin-Stabilized Satellites

## 5. Control System Alternatives

Three basic thermal control systems were studied for SMS application; passive, semiactive, and active. Because the definitions of these systems are not universally accepted they will be restated here. The systems are listed in order of increasing complexity and capability to maintain temperatures within narrow limits.

A passive thermal control system is one in which there are no moving parts and/or fluids affecting the control. In passive systems, temperature control is achieved by using surface coatings with an absorptivity-emissivity combination that will result in the permissible temperature range during all exposures of the spacecraft. Judicious use of insulation, heat paths, radiation barriers, energy storage devices, and control of internal heat dissipation can often supplement the use of surface coatings to control the spacecraft temperature.

A semiactive thermal control system is defined as one containing moving parts and/or fluids, but the system specifically excludes a heat pump. In a semiactive thermal control system heat is rejected at or below the source temperature.

An active thermal control system, by definition, contains a heat pump. In this system, heat may be rejected at a higher temperature than the source temperature. This cannot be done with a semiactive or a passive thermal control system.

There are two basic approaches in passive temperature control design. Since the ultimate heat rejection is through the satellite skin, it is considered good design to place heat generating equipment as close to the outer skin as possible. This will eliminate the extra weight of conductive paths needed to transport the heat from the equipment to the skin. Such an approach will, however, subject the electronic equipment to the same variations in temperature as the skin of the satellite.

Another design approach is to locate the equipment within the satellite interior removed from the outer skin. This design tends to reduce the temperature excursions of the equipment. Here a weight penalty will have to be paid in the provision of sufficient heat paths from the satellite interior to the skin. Analysis shows that due to the low orbital frequency of the SMS, placing heat dissipating equipment in the satellite interior (removed from the skin) will not damp temperature fluctuation appreciably. It is concluded that the first approach is of more advantage for the SMS system.

Each of the three types of stabilization systems (spin-stabilized, three axis stabilized, and gravity gradient stabilized) considered for the SMS affects the thermal control problem in a different way. The spin-stabilized spacecraft, by virtue of its rotation, has virtually an isothermal surface. This reduces the thermal problems considerably. In fact, spin stabilization constitutes an intrinsic semiactive temperature control system. However, since the spinning motion is imparted solely for purposes of attitude control, the system is considered passive from the thermal control point of view. The effect of RPM on the temperature fluctuations along the periphery of a spinning cylindrical satellite is shown in Figure 4-18. It may be seen that even as low

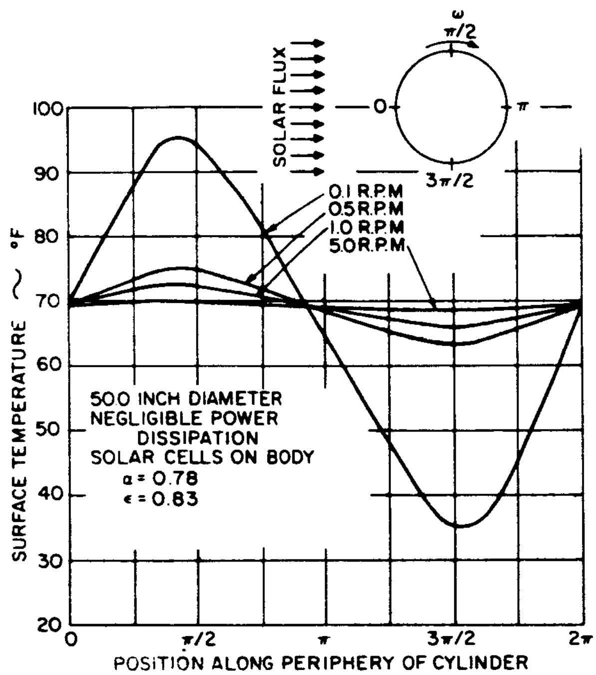



Figure 4-18. Temperature Distribution on a Spin-Stabilized Cylindrical Satellite

an RPM as 0.5 will maintain the surface temperature within close limits. The desirability, from the standpoint of thermal control, of a spin-stabilized spacecraft is evident.

The 3-axis stabilized spacecraft has a heterothermal structure as shown in Figure 4-19. However, total stabilization has one advantage, as far as thermal control is concerned, which a spin-stabilized vehicle cannot offer. This advantage is the possibility of applying the shadow-box technique. Basically, this consists of a self-oriented space radiator with a network of sun-protection grids. This principle and its advantages are discussed in detail in Section 4 of Volume 5. The shadow-boxes, thermal storage material, surface heating elements, and insulation together constitute the passive thermal control system for the 3-axis stabilized SMS.

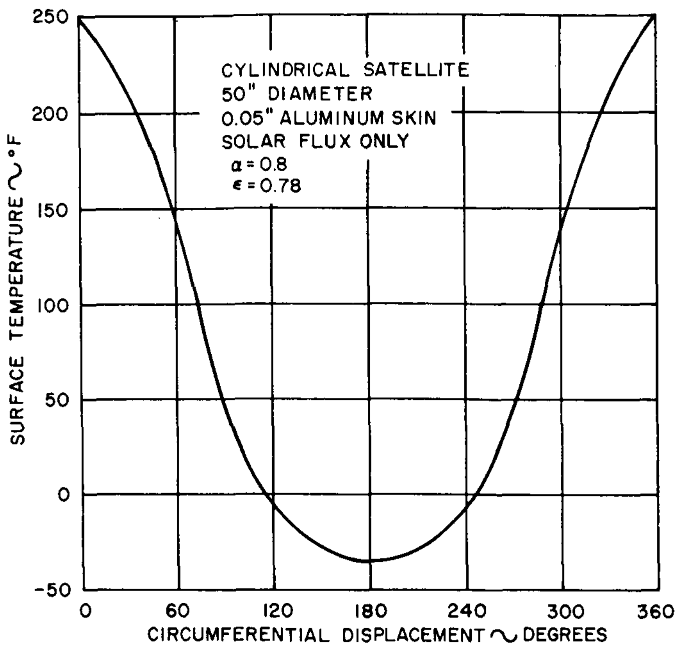



Figure 4-19. Surface Temperature Distribution on a 3-Axis Stabilized Satellite

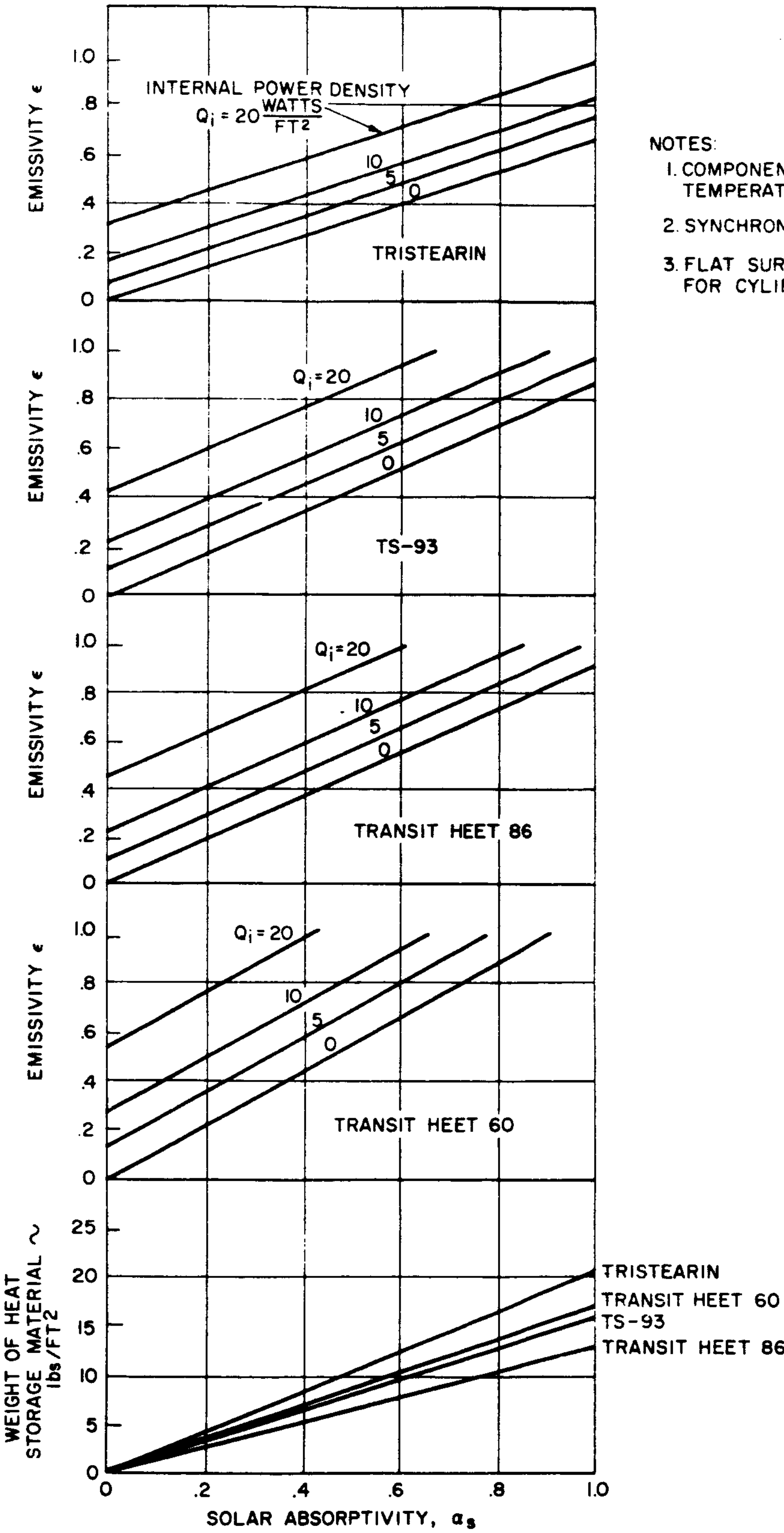
The solution to the thermal control problem of the gravity-gradient configuration is similar to that of the 3-axis stabilized configuration. However, due to the high absorptivity of solar cells (0.7 to 0.8), the possibility of using thermal storage material for temperature control is eliminated. This is due to the prohibitive weight penalty associated with high absorptivities. (See Figure 4-20.)

It is shown that for a solar-cell powered SMS the optimum active thermal control system is one in which the mean radiator temperature is equal to the mean temperature within the equipment compartment. This condition constitutes a semiactive system. Therefore, an active system is not competitive for SMS application. The analysis leading towards this conclusion is presented in detail in Section 4 of Volume 5. Parametric data for passive, semiactive, and active thermal control systems, infrared sensor cooling subsystems, and solar cell thermal protection schemes are also presented.

6. Conclusions and Recommendations for Thermal Control

(a) The SMS 3-axis stabilized spacecraft can be designed so that the thermal control system is entirely passive, i.e., it has no moving parts.

(b) The application of heat storage materials is useful for solving some of the SMS thermal control problems. Laboratory investigation of this material is recommended to demonstrate its characteristics and define the specific material to be used in SMS.




NOTES:

1. COMPONENT TEMPERATURE ≈ FUSION TEMPERATURE.
2. SYNCHRONOUS ORBIT CONDITIONS.
3. FLAT SURFACE. SEE VOL. 2, SECTION 2 FOR CYLINDRICAL SURFACE.

| MATERIAL | HEAT OF FUSION BTU/lb | FUSION TEMP °F | DENSITY lbs/FT2 |
|---|---|---|---|
| TRANSIT HEET 60 | 100 | 60 | 100 |
| TRANSIT HEET 86 | 130 | 86 | 100 |
| TS-93 | 107 | 93 | — |
| TRISTEARIN | 82.1 | 133 | — |

Figure 4-20. Performance of Thermal-Storage Temperature Control Systems

## H. GROUND STATION EQUIPMENT

### 1. General

The ground station system for the SMS would consist of a single command and data acquisition (CDA) station and multiple data acquisition (MDA) stations at numerous locations within view of the SMS. The function of the one CDA station would be as follows:

(1) To receive sensor data from the SMS, and to record it

(2) To receive telemetry data and command signal verification

(3) To transmit commands to the SMS for operation of sensor and for control

(4) To transmit processed meteorological data for relay through the SMS to the MDA stations

The functions of the MDA stations are as follows:

(1) To receive and to record raw sensor data from the SMS

(2) To receive and record processed meteorological data relayed via the SMS

The equipment to perform these functions at the CDA and MDA stations is described in the following paragraphs. Some additional information is available in Sections 4.B and 4.C., on communications in this volume, and more in Volume 5. Not required within the SMS system is ground equipment for launching, for controlling ascent and orbit injection, or for utilization of meteorological data.

### 2. CDA Station

To receive sensor data at 180 MC it is proposed to utilize an 85-ft diameter parabolic antenna. The mount for this antenna can be fairly simple, since steering can be limited to low rates and to small angles. Thus, the antenna cost would be considerably lower than that for a typical steerable antenna. The sensor data receiver would have a low noise front end, consisting of a parametric or helium cooled maser preamplifier mounted close to the antenna. The sensor data receiver would employ frequency modulation with feedback (FMFB) to increase output S/N ratio.

To receive telemetry data at 136 MC during launch and orbit injection, the worldwide Minitrack network would be utilized. When the SMS is on station, a moderate gain (19 DB) antenna at the CDA station could be used to receive telemetry data. This would consist of a four-bay helical type antenna which is commercially available and in use on other programs. The telemetry receiver would be a conventional phase-locked loop type, available commercially.

To transmit command signals at 148 MC to the SMS, it is proposed to use a 5 KW pulse amplitude modulated transmitter of conventional design. The transmitter would either feed an antenna similar to that provided for telemetry reception, or share the same one with a duplexer provided for suitable isolation.

To transmit processed meteorological relay data to the SMS at S-band frequency (2271 MC up and 1705 MC down) a 1 KW transmitter would be utilized. The transmitter would feed the 85-ft antenna mentioned above, with suitable isolation between its two functions. The processed meteorological data might be teletype information, facsimile weather maps or cloud cover pictures. For the satellites having less than 300-lb mass, the command-telemetry systems would be used for a narrowband relay at VHF (148 MC up and 136 MC down). In this case, the processed meteorological relay data would be transmitted from the ground by means of the command transmitter and antenna described previously, sharing time with the command functions.

3. MDA Stations

These stations are designed for low cost and for operation by personnel of medium skill. To receive raw sensor data at 1800 MC directly from the SMS, a modest 30-ft diameter fixed parabolic antenna is proposed. The receiver front end would be a relatively simple (uncooled) tunnel diode preamplifier with a noise figure of 3.5 DB. The difference between receiving sensor data at the MDA stations and at the CDA station is that a 10-DB safety factor would be sacrificed in the former. The same equipment could be utilized for receiving processed meteorological data relayed via the SMS from the CDA station, if the two types of data are sent sequentially rather than simultaneously.

Equipment for command and telemetry is not needed at the MDA stations.

4. Nimbus APT Ground Stations

The Nimbus stations are outside of the SMS system. It was found in studies that relay of processed meteorological data of limited bandwidth through the SMS into these stations can probably be provided, with modest additions or modifications to the station equipment, as mentioned in Section 4.B. Experiments on the existing APT Ground Station equipment are recommended, in order to define the modifications which are expedient or acceptable, and hence whether this relay capability should be required in the SMS.

The study indicated that some modifications will be needed in order to utilize existing Nimbus APT Ground Stations to receive for an SMS relay at 136 MC. Since the SMS is operating at a distance which is one order of magnitude greater than the Nimbus distance, the free space loss is approximately 22 DB greater. To make up this loss, possible modifications would include narrowing the receiving bandwidth and slowing the readout time, slowing the facsimile recorder or changing antennas. The effect of these changes is discussed in Volume 5. It is proposed that experiments be conducted on the existing APT Ground Station to determine what changes could expediently be made for use in the SMS system.

## SECTION 5 - DEVELOPMENT OF ADDITIONAL CAPABILITY

The present parametric study has shown that it is feasible to design and build a synchronous meteorological satellite weighing about 500 lb, in the present state of the art. The problems associated with the design of this spacecraft can be solved by the application of known engineering and design techniques. It is considered, however, that in certain areas it is advantageous to investigate further refinements which have the possibility of improving the final design. The purpose of this section is to delineate major items in this area, as follows:

(1) Cloud Cover Sensors. The sensor tubes recommended at the present time are a Vidicon for the wide-angle picture and an Image Orthicon for the narrow-angle picture. As discussed in Volume 3, other sensors, such as the Image Dissector, SEC Vidicon, Image Intensifier Orthicon, Dielectric Tape Camera, and Ebicon, show promise of meeting the SMS requirements and are under continued development. The use of infrared imaging techniques is also a possibility. The development of these tubes should continue to be monitored and evaluated until the cut-off date, when a final decision must be made for the hardware program to assure that the best possible sensor is used in the SMS.

(2) Attitude Control. A cold gas system is recommended for unloading the reaction wheels. The use of subliming solids for producing the cold gas at 6 psi presents an opportunity of eliminating high pressure tanks and reducing valves, with consequent savings in weight and complexity. This system should be further evaluated for comparison with a high-pressure cold gas system.

There is a control system concept, called the quasi-adaptive rate controller, which would use only cold gas for attitude control, eliminating the need for reaction wheels. Further investigation and demonstration of such a controller would be advantageous in further simplifying the control system.

(3) Attitude Sensors. The use of infrared horizon sensors is satisfactory for the SMS system; however, they are the controlling item insofar as accuracy of attitude holding is concerned. If the development of sensors indicates a need for a closer tolerance on attitude, an improved sensor system, using a star tracker locked on the North Star (Polaris) and a one-gimbal Sun tracker, would provide a highly accurate (on the order of minutes) attitude reference. A horizon sensor system would probably be needed for the original orientation to enable locking on to Polaris and would be available as a backup attitude reference.

(4) Attitude Control Testing. It is usually proposed that a prototype spacecraft be tested on an air bearing in a space chamber. For the synchronous orbit, however, the largest disturbance torque is on the order of 0.000005 in/lb, while the best air bearing will have a torque of about 0.05 in/lb, which can mask the results if the actual value of disturbance torques is used in tests. An alternate method of testing would be to mount the spacecraft torque-producing devices on force balances in the space chamber, along with the prototype SMS. A computer would then be used to simulate the dynamic response of the spacecraft. This combination would thus test flight hardware in a near vacuum. Such a system would eliminate the following problems:

(a) Deflection of the vehicle when mounted on an air bearing, which can introduce significant errors

(b) Restriction on angular rotation, as the spacecraft does not move; the rotation response is supplied by the computer

(c) Compromise of the spacecraft design because of the need to provide access to the cg for mounting the spacecraft on the air bearing. It is recommended that the test procedure be determined early in the design phase as it has a direct influence on the structural configuration.

(5) Thermal Control Material. It has been proposed that thermal storage material be used in the SMS as part of the passive thermal control system. This material has been used with success in ground applications but not in spacecraft. It is suggested that a small laboratory investigation be conducted early in the design phase in order to select the specific heat storage material which will best meet the SMS requirements and to test its design application.

(6) Flexible Solar Cell Paddles. The present design is based on the use of hinged panels on which the solar cells are mounted. These can be fitted within the launch vehicle nose shroud. The use of a flexible mounting surface would permit better stowage; and, more important, it would tend to alleviate the severe vibration problems which occur with large rigid panels. Investigation of flexible solar panels for application to the SMS is recommended.